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Development and Testing of the Propulsion Subsystem for the Mariner Mars 1971 Spacecraft

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PREFACE

The work described in this report was performed by the Propulsion Division of the Jet Propulsion Laboratory, under the cognizance of the Mariner Mars 1971 Project.

ACKNOWLEDGMENTS

This report was assembled from documentation authored by many JPL personnel and subcontractors; the report was compiled and edited by R. R. Breshears, R. D. Cannova, M. J. Cork, and D. D. Evans. Other personnel contributing to this effort include E. Cuddihy, C. Dodge, W. Dunn, S. Epstein, A. Giandominico, P. Gordon, L. Jones, L. Mattson, T. Metz, R. Oshiro, D. Phillips, M. Pompa, D. Schmit, H. Stanford, J. Stocky, R. Warren, R. Weiner, and G. Yankura of JPL, and E. Moser and L. Norquist of Martin Marietta Corporation. Material was included from the two major subcontractors -- Martin Marietta Corporation, Denver Division, contractor for the propulsion feed system, and North American Rockwell, Rocketdyne Division, contractor for the rocket engine assembly.

For purposes of reference and retrieval, the authorship is given as "Richard D. Cannova et al."

Values in customary units are included in parentheses after values in SI (International System) units if the customary units were used in the measurements or calculations.

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ABSTRACT

In November 1971, the Mariner 9 spacecraft was injected into Martian orbit by a 574-kg (1265 lb_m) propulsion system. Design of that system provided directed impulse, upon command, to accomplish in-transit trajectory corrections, an orbital insertion maneuver at encounter to transfer from a flyby to an orbiter trajectory about the planet Mars, and subsequent trim maneuver.

The propulsion system is an integrated, pressure-fed, multi-start, fixed thrust, storable bipropellant system. The primary subassemblies are a propellant feed system, a 1334-N (300 lb_f) thrust rocket engine assembly, and the propulsion module structure. The subsystem was capable of being fueled, pressurized, and monitored before installation on the spacecraft.

This document describes the design, testing, fabrication, and problems associated with the development of the Mariner 9 propulsion system. Also covered are the design and operation of the associated ground support equipment used to test and service the propulsion system.

I. INTRODUCTION

The Mariner Mars 1971 (MM '71) project was started in 1968 with the object of orbiting a spacecraft around the planet Mars. Early conceptual designs of the propulsion subsystem drew heavily upon previous system designs and the use of qualified components.

The Mariner 1971 program is a direct follow-on to the Mariner 1969 program and relies to a great extent upon the Mariner 1969 spacecraft design. However, since the Mariner 1971 spacecraft was to orbit Mars (rather than fly by), a larger propulsion subsystem was required.

The work reported here describes the development of the propulsion subsystem. The analyses and interface aspects of the subsystem are described in Ref. 1, and the management aspects are reported in Ref. 2.

The function of the propulsion subsystem is to provide directed impulse, upon command, to accomplish in-transit trajectory corrections, an orbital insertion maneuver at encounter to transfer from a flyby to an orbiting trajectory about the planet Mars, and subsequent trim maneuvers.

The propulsion subsystem is an integrated, pressure-fed, multistart, fixed-thrust, storable bipropellant system. Nitrogen tetroxide (N_2O_4) and monomethyl-hydrazine (MMH) are used for propellants, and gaseous nitrogen is used for pressurization.

The primary subassemblies are a nitrogen tank supply, a pressurant control assembly that provides pressurant isolation and regulation, two check and relief valve assemblies, two propellant tanks with positive expulsion bladders and gas trap standpipes, two propellant isolation assemblies, a gimballed 1334-N (300 lb_f) thrust rocket engine assembly with an electrically operated bipropellant valve, and the propulsion module structure. The propulsion subsystem is shown schematically in Fig. 1. Nitrogen pressurant is isolated from the remainder of the subsystem by the pyrotechnic valves of the pressurant control assembly (PCA). Upon actuation of one of the PCA normally closed valves, pressurant is permitted to flow from the pressurant tanks through both the 12-μm absolute pressurant filter and the regulator, whose outlet pressure is controlled to be $1758 \times 10^3 \pm 55 \times 10^3$ N/m^2 (255 ±8 psi) above local ambient pressure. Pressurant then flows into the pressure check and relief assemblies (PCRA), where check valves prevent backflow of propellant saturated pressurant, and on to the propellant tanks.

Once in the propellant tank, the pressurant causes the bladder to collapse around the standpipe and expel propellant through the gas retention device in the standpipe and into the propellant isolation assembly (PIA). The PIA is used to control propellant flow to the rocket engine with three normally closed and two normally open pyrotechnic valves and to filter this flow with a 35-μm absolute propellant filter. After leaving the PIA, the propellant flows through flex lines to the rocket engine; the flex lines permit gimballing of the rocket engine.

Servicing valves are used to provide access to the inlet and outlet sides of the pyrotechnic valves in the PCA and PIAs, to the downstream side of the check valves in the PCRA, and to the propellant tank side of each standpipe. Pressure transducers provide pressure information at the PCA inlet, downstream of the check valves in the PCRA, and the PIA outlet.

The rocket engine operates with a mixture ratio of N_2O_4 to MMH mixture of 1.57/1; the hot gases are expelled through a nozzle with an expansion ratio of 40:1.

The propulsion support structure, a beryllium tube truss with magnesium and steel fittings, is attached to the upper octagonal ring and supports the propulsion equipment, the high-gain antenna, and the low-gain antenna.

The propulsion subsystem is capable of being fueled, pressurized, and monitored before installation on the spacecraft. At launch, the propellants and high-pressure gas supply are isolated by the pyrotechnic valve assemblies. Before the first trajectory correction, the engine valve is opened to bleed the air trapped between the normally closed propellant pyrotechnic valves and the engine valve. Actuation of the first set of pyrotechnic valves P1, F1, O1, (Fig. 1) allows the propellant tanks to be pressurized and allows propellant to flow to the engine valve. The trajectory-correction maneuver is then performed by opening the engine valve, thus allowing the propellants to flow into the thrust chamber, to undergo hypergolic ignition, and to continue to burn until such time as the desired velocity increment is obtained. At this time, the engine valve is closed by removing its electrical power. When tracking data confirms that no more propulsion maneuvers will be required before the nominal time of the second trajectory correction, the propellant and pressurant lines are closed by actuation of the second set of pyrotechnic valves to guard against leakage. The pressurant and propellant lines are reopened, by the third set of valves, just before the second trajectory correction, if required. This set of valves remains open for the time period of the orbit insertion and initial trim maneuvers. The orbit insertion maneuver consists of a burn of approximately 900-s duration to place the spacecraft into planetary orbit. One or two short-duration trim maneuvers are required to refine the orbital parameters. After tracking data confirm correct orbital characteristics, the fourth set of valves provides the capability to isolate the propulsion fluids for the remainder of the mission. A fifth set of valves is available to be used (1) in case one of the other valves fails to open, or (2) to open the system late in the mission to perform special maneuvers.

The propulsion subsystem is shown on its handling fixture in Fig. 2.

Actuation of the pyrotechnic valves and management of solenoid power for the engine valve are accomplished by power switching in the pyrotechnics subsystem. Thrust vector control during engine firing is provided by the use of gimbal actuators for pitch and yaw control and cold gas jets for roll control.

II. PROPULSION SUBSYSTEM CHARACTERISTICS AND PERFORMANCE

A. Requirements

The following design requirements for the propulsion subsystem are a result of spacecraft physical and operational constraints, launch vehicle characteristics, ground and in-flight environmental conditions, and propulsion subsystem characteristics. The propulsion subsystem is required to:

(1) Be capable of carrying 440 kg (970 lb_m) of usable propellant (total load less unusable residuals) with a propellant specific impulse of 2775 ±49 m/s (283 ±5 lb_f-sec/lb_m).

(2) Be capable of providing a minimum impulse of 5338 N-s (1200 lb_f-s) for the first midcourse maneuver and 534 N-s (120 lb-s) for any subsequent maneuver.

(3) Be capable of five starts and thrust terminations in a vacuum and gravitationless environment with a minimum time between consecutive engine firings of 24 h.

(4) Be capable of a dry and unpressurized storage life over a temperature range of 10 to 32°C (50 to 90°F) for at least 18 months.

(5) Be capable of serviced life over a temperature range of 15 to 32°C (+60 to 90°F) at less than 70% relative humidity for at least 2 months with two transient exposures to air at 40°C (105°F) and 90% relative humidity, provided that the average propellant temperature in either tank does not exceed 32°C (90°F).

(6) Function without degradation of performance after vacuum and gravitationless environment exposure over a temperature range of 0 to 32°C (+30 to 90°F) for a period of 300 days after launch.

(7) Exhibit external leakage rates for the duration of the mission no greater than 5.5×10^{-3} STP cm^3/s of gaseous nitrogen, except for possible relief valve venting.

(8) Generate torques through the relief valve vents of no greater than 0.17 m-N (1.5 in.-lb_f) in roll and 0.056 m-N (0.5 in.-lb_f) in pitch and yaw in the event of regulator failure.

(9) Weigh no more than 98 kg (216 lb_m), exclusive of primary structural elements, cabling, gimbal actuators, squibs, thermal shields, and propellant.

(10) Have a propellant load capacity of not less than 462.7 kg (1020 lb_m).

(11) Consume no more than 35 W of electrical power at 30 Vdc during engine operation for midcourse and orbit trim maneuvers (no more than a 30-W average for orbit insertion) and consume no power during non-operational modes, except for temperature control heaters as required.

B. Interfaces

Typical interfaces with the propulsion subsystem and the spacecraft are given below.

1. Propulsion/structure subsystem interface. The propulsion subsystem assembly is mounted and aligned to the spacecraft structure subsystem to allow the propulsion subsystem to be separated from or assembled to the spacecraft bus in either a dry or a fully loaded condition.

2. Propulsion/flight telemetry subsystem interface. Requirements are as follows:

(1) Propulsion subsystem pressure transducers are supplied with a regulated excitation voltage of 3 ±0.1 Vdc.

(2) Propulsion subsystem telemetry outputs consist of the measurements shown in Table 1.

3. Propulsion/pyrotechnic subsystem interface. A continuous electrical signal at $30^{+1.5}_{-2.0}$ Vdc, under load, will be provided from the pyrotechnic subsystem for operation of the engine valve during propulsion maneuvers.

4. Propulsion/attitude control subsystem interface. Thrust vector control will be provided by the attitude control subsystem as follows: (1) pitch and yaw control by means of gimbal actuators attached to the rocket engine assembly and (2) roll control by means of cold gas jets.

C. Propulsion Performance

Propulsion subsystem performance parameters are presented in Table 2. The method of performance determination during the Mariner 1971 program consisted of three primary elements as listed below:

(1) Estimate and measure subsystem dry mass during the course of the development and qualification program.

(2) Select a nominal operating region in which the subsystem would perform reliably while meeting performance requirements.

(3) Determine optimal propellant loaded mixture ratio to maximize performance capability.

1. Subsystem mass. Propulsion subsystem dry weight estimates evolved from purely analytical estimates to a group of measured component weights with calculated connections to a measured assembly weight. The weight allocation of 98 kg (216 lb_m) (see Sec. II-A) represented 14 kg (31 lb_m) N_2 and 84 kg (185 lb_m) dry weight; this value was negotiated at a time when some component weights were actual but most were analytical predictions. The allocation was not considered a dynamic limit but rather a static guideline during the remainder of the program. Little emphasis was placed on weight reduction because the program was cost limited and because launch vehicle and spacecraft propulsion performance commitments improved enough during the program to offset increases in dry weight.

Propellant tank size for the propulsion subsystem was determined in mid-1968. The selected size of 221,265 cm^3 (13,500 $in.^3$) per tank represented about 10% margin (at nominal 1.55 mixture ratio (MR) over the 435-kg (960-lb_m) propellant load required at that time. This margin was included because of uncertainties in spacecraft mass, velocity increment required, effect of propellant N_2 saturation on MR, and engine thermal margin. The first two items could cause an increase in total propellant load, and the latter two could cause a decrease in loaded MR and total load capacity.

2. Operating performance. Engine, feed system, and subsystem development and margin-limit tests were conducted to characterize performance parameters. These parameters included engine thermal margin, N_2 saturation rates, N_2 saturation effects on MR, propellant tank expulsion efficiency, and uncertainties in operating conditions. The results of these tests led to the following strategy for determination of propellant loads and flight subsystem performance predictions:

(1) Calibrate type approval (TA) and all flight engines to a target MR of 1.57 at rated inlet conditions of 1634×10^3 N/m^2 (237 psia) oxidizer and 1668×10^3 N/m^2 (242 psia) fuel.

(2) Load all systems with a maximum fuel load of 186 kg (410 lb_m)(4% ullage at 21°C (70°F).

(3) Load the TA subsystem with a nominal oxidizer load of 288 kg (635 lb_m). The flight subsystem load could be expected to vary slightly as a function of nominal engine MR, TA subsystem test results, and/or average N_2 saturation level predicted for the reference mission profile.

3. Performance predictions. Table 3 is the final component-level dry-mass summary for the propulsion subsystem. Also listed are the items provided by other subsystems to complete the propulsion assembly. Measured assembly weights corresponded to the value given within the weighing accuracy of 0.9 kg (±2 lb_m).

A nominal pressure budget and performance summary for the TA propulsion subsystem at rated conditions (21°C or 70°F, unsaturated propellants) is presented in Table 4. This summary was calculated with a system-balance computer program after the engineering test model (ETM) and TA subsystem test results had been analyzed. The test programs showed that dissolved gas does not effect engine operating characteristics until the partial pressure of nitrogen in solution exceeds the injector inlet pressure. Fully saturated oxidizer (at 1758 N/m^2 (255 psi) tank pressure) causes an MR decrease of about 0.073 MR units, while fuel saturation causes a 0.016 increase in MR. A 15°C (60°F) increase in temperature 0 to 32°C (30 to 90°F) was seen to decrease fuel-side resistance by 1.6% and oxidizer-side resistance by about 1%.

A performance summary for the Mariner 9 propulsion subsystem is presented in Table 5. Performance optimization for a reference mission profile resulted in a 290-kg (640-lb_m) oxidizer load. The nominal MR for the Mariner H spacecraft was 0.02 MR units less, so that 287 kg (633 lb_m) of the oxidizer was loaded.

Note that the final propellant mass estimates of Table 5 are made up of two components. The performance reserve represents an rss increase of propellant usage from nominal, due to 3σ preflight uncertainties in specific impulse, engine MR, feed system ΔP, propellant saturation level, propellant temperature, and engine-valve-seat coining. This allocation is weighted toward the fuel side because the saturation effect is predominantly one-sided toward low MR. The oxidizer holdup of 6.6 kg (3 lb) was allocated for vapor permeation through the bladder. The remainder of the oxidizer and all of the fuel holdup corresponds to 410 cm^3 (25 $in.^3$) line volume, 3606 cm^3 (220 $in.^3$) standpipe internal volume, and 2950 cm^3 (180 $in.^3$) bladder holdup. Most of the bladder holdup could be expelled if bladder ΔP were allowed to increase from 34.48×10^3 N/m^2 (5 psid) to 1724 N/m^2 (250 psid). The combined holdup allocation of 6966 cm^3 (425 $in.^3$) represented about 3.1% of tank and line volume.

The resulting ΔV available if propellant were depleted to both oxidizer and fuel minimum values would be 1683 m/s as shown in Table 5. This value could decrease in flight if a different mission profile, hence different average saturation levels, were flown rather than the one used for performance optimization. However, knowledge of the subsystem performance during orbit insertion, as obtained from spacecraft telemetry, may allow a decrease in the performance reserve requirements and a resultant increase in performance commitment. It is expected that the final performance commitment will be between 1670 and 1700 m/s. This provides some margin over the 1650-m/s requirement. Final flight performance is reported in Ref. 3.

III. DEVELOPMENT

Subsystem level development of the MM '71 propulsion subsystem was performed through the fabrication and testing of three operating subsystems -- development system model, engineering test model, and type approval subsystem. These represent the counterparts of breadboard, prototype, and type approval subsystems in electronic subsystem terminology.

A. Development System Model

A development system model (DSM) was constructed to obtain early knowledge of system operation and interactions between components. To obtain preliminary data on the system, two approaches were used. The conceptual design was evaluated by the DSM, which was built of surplus hardware in the general operating range. Components which had the proper characteristics were also procured and evaluated. As components completed evaluation, they were incorporated into the DSM.

The initial DSM configuration included propellant tanks, bladders, and pressurant tanks from the Gemini program, a modified gas regulator from the Minuteman program, and check valves from the Lunar Orbiter program. Other components were from earlier JPL programs or were development items for MM '71. The subsystem was assembled on a test frame for atmospheric testing at Edwards Test Station (ETS). Instrumentation was installed in addition to the planned flight parameters to better define subsystem performance. DSM Tests 1, 2, and 3 were conducted in May and June of 1969, as indicated in Table 6.

The DSM was rebuilt after Test 3 to include flight geometry heavyweight propellant tanks with flight-type bladders and heavyweight standpipes; most of the other components were flight-type. Diameter, length, and volume of pressurant and propellant tubing were made as flight-like as possible. The subsystem was installed in the new vacuum chamber at Dv stand at ETS and tested through Tests 4, 5, and 6 of Table 6.

The DSM tests demonstrated satisfactory, confidence-building results regarding facility operating procedures, instrumentation, propellant loading and unloading, regulator operations, engine-feed system interactions, engine performance, and effects of nitrogen-saturated oxidizer on engine mixture ratio. The need for further work was indicated regarding the effect of N_2 outgassing from saturated oxidizer engine start transients, the ability of standpipe and propellant line pressure transducer to withstand higher than expected feedline pressure surges, and the definition of pressure-temperature characteristics of the closed volumes between valves O2-F2 and the engine valve during interplanetary cruise.

Design changes required were (1) replacement of 0.635-cm (1/4-in.) tubing by 0.952-cm (3/8-in.) tubing between the regulator and propellant tanks to prevent overpressurization during regulator slam start and (2) addition of a bypass connection in the gas side of the propellant tanks to allow depressurization of the propellant tanks in the launch (engine-up) configuration. (The Teflon bladders being acted on by liquid pressure-head prevent venting through the primary gas inlet port.)

B. Engineering Test Model

The MM '71 engineering test model was a flight prototype configuration propulsion subsystem (PSS) which was intended to provide extensive information relative to the MM '71 PSS. It was to serve as a test system to evaluate operation and performance of the subsystem over a wide range of conditions and environments. It was also to serve as a pathfinder for fabrication, assembly, checkout and other operational aspects of the flight subsystems. Details of the ETM are given in Ref. 4.

In December 1969, assembly of the ETM was started. In general, the flight procedures for assembly and brazing were utilized. The ETM PSS then underwent proof test, external leakage test, functional test, and vacuum chamber leak test, utilizing the flight procedures. All tests were satisfactory, except that both check valves indicated out-of-specification leakage.

The ETM was delivered to Edwards Test Station, loaded with solvents, and then subjected to flight approval (FA) and TA vibration in the Z axis. In Fig. 3, the ETM is shown on the vibration fixture. A structural failure occurred in the tank support struts, which resulted in a basic change in the ETM plans.

The ETM struts were replaced with aluminum struts (from a mockup system), the ring frame was repaired, and the PSS was subjected to axial loading test to confirm structural integrity. Also, a bladder leak had developed in the oxidizer tank. A new bladder and a flight standpipe were installed in the oxidizer tank at ETS. The ETM was subjected to another proof test, external leak test, and functional test, and all components showed no degradation. In fact, the check valves indicated zero leakage. The ETM was then rotated and installed in the vacuum chamber (stand DV) on May 15, 1970. The basic objective of propellant vibration to simulate the boost phase environment was cancelled due to the structural failure.

Five hot-firing test series were conducted on the ETM to (1) simulate various mission duty cycles, (2) evaluate performance and operation over a wide range of conditions, (3) serve as a pathfinder for the conditions planned for the TA program, and (4) evaluate performance after long-term (3-month) exposure to propellant.

The results from the five test series, Tables 7 and 8, indicated that the ETM operated and performed very satisfactorily. No major failures or difficulties were encountered. Performance, in general, was near predictions, although it was found through the data analysis that adjustments to the predictions were desired. All components functioned properly, and satisfactory reliability was demonstrated. It was demonstrated that the PSS would operate properly and predictably when exposed to conditions beyond those expected in the actual flight. Examples of these conditions are: (1) high propellant temperatures (32°C or 90°F), (2) low propellant temperatures (6°C or 44°F), (3) fully saturated propellant, (4) pyrotechnic valve opening with high tank pressures, (5) temperature increase after pyrotechnic valve closing, (6) exhaustion of GN_2 pressurant, and (7) near-complete propellant expulsion (fuel side).

Specific results of the ETM tests are as follows:

(1) General. No external leakage was noted at any time, including a 3-month storage period.

(2) Pressurant feed system.

(a) The regulator regulated within 27.5×10^3 N/m^2 (4 psi) of the acceptance test data, and there was no indication of regulator leakage throughout the program.

(b) Depletion of gas had no effect on operation, other than to cause the propellant tanks to operate in a blowdown mode, with a corresponding decrease in thrust.

(c) The check valves initially leaked on assembly; however, after vibration, no leakage was noted. After 3 months exposure, the oxidizer check valve cracking pressure was 4.1×10^3 N/m^2 (0.6 psi) above nominal; however, fuel check valve cracking pressure was nominal. Five test series did not significantly affect check valve operation.

(d) Burst disks and relief valves operated satisfactorily (relief cracking had increased slightly) after 3-month exposure.

(3) Propellant feed system

(a) Propellant saturation rates, over a long time period (3 months) were approximately near expected values.

(b) A 98.7% propellant expulsion on the fuel side was satisfactorily achieved, which resulted in a slight decrease in engine chamber pressure just prior to shutdown. A 97.4% propellant expulsion was achieved with no decrease in engine chamber pressure. Five expulsion cycles ranging from 92% to 98.7% were conducted on the fuel bladder (codispersion) with no significant change in bladder leakage.

(c) The oxidizer bladders experienced expulsions ranging from 84.7% to 96.9%. The laminate bladder experienced liquid leakage after four expulsion cycles.

(d) Feedline pressure surges did not affect gas retention of the oxidizer standpipe (the fuel standpipe did not have screens).

(e) The fuel injector pressure transducer experienced a much greater calibration shift (1130 N/m^2) (164 psi) than would be expected in flight as a result of worst-case pyrotechnic valve opening transients. No other damage was apparent.

(f) When the liquid pyrotechnic valves were closed, increasing temperatures caused an increase in pressure in the trapped liquid at the rates predicted.

(4) Engine performance

(a) Engine hydraulic resistance did not change through four simulated missions. Pretest and posttest solvent flow calibration data were identical.

(b) The actuators controlled gimballing of the engine satisfactorily, and there was no evidence of excessive loads or abnormal operation.

(c) The engine valve operated satisfactorily, with no evidence of out-of-specification leakage, through the complete test program, which included exposure to boiling oxidizer after a high-temperature engine test.

C. Type Approval Test Model

The broad objectives of the TA program were, as nearly as practical, to simulate the processes, interfaces, tests, environments, and mission that an actual flight subsystem would experience. In addition, it was intended to expose the subsystem to limits or environments, where appropriate, beyond expected conditions, so as to demonstrate a level of margin. Most notedly, the extended conditions were (1) higher level and increased duration for mechanical vibration, (2) operation at extreme temperature limits, (3) two mission duty cycles, (4) extra handling and servicing, (5) additional functional and component checks, and (6) other extended operating limits such as high tank pressures, extreme nonoperating temperatures, and extreme Moog valve temperatures.

The TA subsystem, by definition, was intended to be of flight configuration. This implies exposure to acceptance tests and other processes that the flight equipment would be exposed to. In general, this was true; significant deviations are listed below:

(1) Both propellant tanks contained laminate TFE-FEP bladders rather than the flight configuration codispersion bladders. A penalty TA program was conducted later to qualify the new bladder (see Sec. IV-I-10).

(2) The rocket engine assembly had some deviation in the gimbal ring mounting bracket bolts, which included higher torque, coarse threads, less spacer bearing area, and no lockwire provisions. Also, the injector boundary layer cooling injector ring weld was accomplished with higher beam current and excessive penetration was suspected. These later deviations were substantiated after the completion of both test series, in that streaking on the chamber was noted, and two injector orifices were determined to be plugged by contaminants created by excessive weld penetration.

(3) The structure was different from the flight configuration in three areas:

   (a) The pressurant tank bipod support tubes were changed from beryllium to steel after Z-axis vibration with solvents.

   (b) The nitrogen tank support fitting was redesigned for flight and was incorporated on the PSS after the hot firing series.

   (c) Ring frame doublers were added to the flight units and were incorporated after the hot firing series.

The TA subsystem was assembled, tested, handled, and in general exposed to conditions similar to those that flight units would experience. Following this, the TA subsystem underwent two simulated mission duty cycles. In the vibration testing, the TA was subjected to more severe conditions than expected on the flight units, and in the two mission duty cycles, the TA unit was exposed to specific extended environments.

The first extended condition for the TA PSS was Z-axis vibration with solvents. The FA level was conducted to simulate the flight acceptance subsystem test to be conducted on the PTM and flight units. However, in addition, the flight units experience FA spacecraft-level vibration, and the PTM subsystem experiences spacecraft-level TA vibration. In order to test for this added spacecraft-level vibration, the TA subsystem was subjected to TA-level vibration in the Z-axis.

The second extended condition imposed was TA-level vibration in three axes. This test was conducted to demonstrate margin over the launch and boost phase vibration environment, with the vibration levels higher than expected for flight, and with longer exposure times.

Two simulated mission duty cycles were also conducted, as outlined in Tables 9 and 10. The test events were planned to simulate the flight events insofar as possible. The TA subsystem is shown in Fig. 4, mounted in the vacuum chamber facility.

One problem which had an implication relative to flight operation or reliability was identified during the TA test program. This problem was a high cracking pressure on the oxidizer check valve during the first propellant tank pressurizing cycle and first engine burn. Should this same anomaly occur in flight, on the orbit insertion maneuver, and the valve not correct itself during the burn, calculations indicate that a nominal mission could be achieved, but nearly all performance margin would be lost (refer to Sec. IV-F-3).

Reference 5 documents the activities, data, analysis, and other information gathered during the PSS TA program.

The TA subsystem operated and performed as predicted or expected. All components functioned as expected, except for the oxidizer check valve and the pressurant tank bipod. All specification requirements were satisfied. Reliability was demonstrated and all interfacing equipment such as pyrotechnic, thermal, structure, support equipment, and test facilities operated and functionally interfaced satisfactorily.

IV. PROPELLANT FEED SYSTEM

The fabrication of the propellant feed system major subassemblies, as identified in Fig. 1, was performed by the Martin Marietta Corporation, Denver Division (MMC) under contract to JPL. This responsibility included the procurement of the components and their acceptance and qualification testing. The only components not purchased by MMC were the propellant tank shells and the flex lines, which were procured by JPL. The components were incorporated with detail parts machined by MMC to form the subassemblies, which were then acceptance tested and provided to JPL.

Upon their receipt at JPL, the subassemblies were mounted on the subsystem structure and joined to their interconnecting plumbing. When assembly of the propulsion subsystem was completed, it was then subjected to the subsystem flight acceptance test.

The connection of components within subassemblies and the interconnection of subassemblies within the propulsion subsystem was accomplished by brazing (see Sec. VI). With this technique the number of mechanical external seals on the subsystem were reduced to sixteen: ten service valves, each with a primary and a redundant seal; two tank flanges with aluminum crush gasket seals; and four "AN-type" fittings, two on each flex hose, with crushable aluminum seals. This fabrication technique resulted in a subsystem external leakage rate of less than 1×10^{-5} STP cm^3/s when the subsystem was pressurized to its operating pressures with helium.

The experiences with the feed system portion of the propulsion subsystem permit the following conclusions:

(1) The use of an induction brazing process resulted in a generally leakage-free assembly. This process, however, was found to be a contamination source, and protection should be provided for contamination-sensitive components.

(2) The use of identifiable groups of components as separate subassemblies to be treated at the subsystem level as modules resulted in a simplification of the design, fabrication, assembly, and test of the feed system.

Components from existing programs were selected wherever possible to minimize development and qualification. Some minor changes and improvements were incorporated in several components due to the subsystem requirements and long-term exposure to propellants. Detailed documentation of feed system component and subassembly test programs is given in Ref. 6. The following sections summarize component functional characteristics, JPL test activities, and problems encountered.

A. Pressurant Tank

The MM '71 pressurant tank is a fully annealed 6A14V titanium vessel machined from hemispherical forgings and is joined at an equator by a tungsten inert gas (TIG) weld. It has a nominal operating pressure of 27.6×10^6 N/m^2 (4000 psig), a room temperature proof of 41.4×10^6 N/m^2 (6000 psig), and a minimum burst pressure of 55.2×10^6 N/m^2 (8000 psig). The specified minimum ultimate strength for the annealed material is 930.8×10^6 N/m^2 (135000 psi), and minimum yield is 861.8×10^6 N/m^2 (125000 psi). The minimum allowable wall thickness was established at 0.569 cm (0.224 in.) for the 37.1-cm (14.6-in.) ID, 26614-cm^3 (1624-$in.^3$) volume tank. The connection of the titanium tank to the stainless steel plumbing is by means of a 0.635-cm (1/4-in.) stainless steel/titanium transition tube welded to a 0.635-cm (1/4-in.) titanium tube, which is, in turn, welded to the single tank opening drilled through the upper mounting tab. The tank tabs match fixed locations on the spacecraft.

The only problem associated with pressurant tank development was with the titanium-to-stainless-steel transition tubes. Figure 5, which is a schematic of the pressurant tank, shows the relative location of the transition tube.

Microsectioning the transition tubes on several test tanks for a close look at nonmetallic surface inclusions revealed separation at the explosively formed bond. Metallurgical evaluation of several tubes removed from the pressurant tanks confirmed the presence of joint separations of up to 70% of the length of the interface. In all instances, the bond joint separation was most severe at the tube ID, with effective bonding occurring only at the outer portion of the tube wall. Although the specific cause of the separation was not confirmed, it was suspected that the problem had its origin in the explosive forming process, which was the method used to fabricate these small-diameter transition tubes.

A similar metallurgical examination of tubes of the same size and wall thickness (0.635-cm) (1/4-in. OD) X (0.02629-cm) (0.0135-in. wall) that were fabricated by a coextrusion method indicated that this method produced transition tubes with satisfactory bond joints. The original tank fabricator (Fansteel Advanced Structures Division, Compton, California) was contracted to remove the explosively formed transition tubes from seven tanks and replace them with coextruded tubes.

B. Pyrotechnic Valves

The pyrotechnic valves (designed, manufactured, and tested by Pyronetics, Incorporated, a subsidiary of The Cosmodyne Corporation, Torrance, California) are 1.27-cm (1/2-in.) valves developed for this program. The design principles were derived from similar 1.27-cm (1/2-in.) valves used on other programs and from the 0.635-cm (1/4-in.) valves used on previous Mariner flights.

Each valve is actuated by the explosive energy generated by a single MM '71 standard pyrotechnic cartridge. The motion imparted to the ram by the cartridge gases shears a section of the parent metal in the flow passage, which either initiates or terminates flow with a minimum of metallic or cartridge gas contaminants injected into the flow passage.

The initial seal between the combustion chamber and the flow passage consists of the primary unsheared wall of the nipple and a secondary Bal Seal (a Teflon U-shaped seal internally backed up by a stainless spring) mounted in the ram (Figs. 6 and 7). The nipple wall is sheared when ram motion is initiated, and the Bal Seal becomes the primary dynamic seal between the cartridge gas and the flow passage. The ram energy is dissipated by deformation of the valve body bore at the end of the stroke, thus effecting the final metal seal as indicated in the figures.

An early test program was established to verify valve/cartridge compatibility. Six test firings were performed using prototype cartridges in three production valves of each type exposed to temperatures of -54 and +56°C (-65 and +125°F) at pressures of 1620 X 10^3 N/m^2 (235 psig) and 27.6 X 10^6 N/m^2 (4000 psig), with GN_2 and water as the pressurizing media. The data indicated normal cartridge/valve performance.

Twenty-four normally open and 24 normally closed valves were subjected to the qualification test program listed in Table 11. These valves also served the dual purpose of Lot No. 1 acceptance. All test valves completed the test program satisfactorily.

C. Pressure Regulator

The pressure regulator was fabricated by the National Water Lift Company (a Division of Pneumo Dynamics Corporation), of El Segundo, California. The regulator has a single stage and uses a hard seat ball poppet valve with pneumatic and friction damping. This design automatically maintains constant outlet pressure by means of a spring loaded, vented metal bellows. The unit is shown in cross-section in Fig. 8.

The regulator maintains constant outlet pressure by self-positioning of the ball poppet in such a way that flow is increased when the outlet pressure is less than the regulator set point, and decreased when greater. The poppet motion is controlled by the capped bellows, which is deflected by any imbalance between the force exerted by the outlet pressure on one side, and the forces due to the reference springs and the ambient pressure on the other. The regulation set point is therefore determined by the reference springs and the ambient pressure.

The regulator contains a corrugated wire screen filter rated at 10 μm absolute in the inlet side. The filter is preceded by a venturi sized to limit flow through the unit to 327.6 STP cm^3/s (195 STP ft^3/min) GN_2 at 16°C (60°F) or less at 27.6×10^6 N/m^2 (4000 psig) inlet pressure. All sliding surfaces are Teflon-to-metal and no lubricants are used. All external joints that carry the structural loads are screw threaded and seal welded.

Functional testing of an engineering development unit in a mockup of the original subsystem design revealed the following problems:

(1) Instability at low flow rates. This problem was eventually shown to be due to the check valves. The instability was measurable but was within the pending specification limits and occurred well out of the nominal flow rate range.

(2) Excessive overshoot in outlet pressure during slam start events when the subsystem mockup was made with 0.635-cm (1/4-in.) tubing. The subsystem line size was changed to a 0.476-cm (3/8-in.) tube is a result of this test and DSM subsystem tests (Sec. III-A).

No further problems that indicated failure, deterioration, or marginal design performance were observed during the course of regulator development and type approval testing, as documented in Ref. 6. Two assembly problems that did occur are discussed in the following paragraphs.

A pressure regulator failed to lock up during flight acceptance testing of the first pressurant control subassembly at MMC. The internal leakage rate was estimated to be 15.28 STP cm^3/s (0.0025 lb/s). The failure did not repeat after the regulator was operated again. It was concluded that the excess internal leakage was caused by contamination which was caught between the poppet and seat and blown out during the subsequent operation of the regulator. An alternative consideration was that an ice film formed on the seat from "wet" GN_2 and prevented the poppet from sealing. This theory has merit in light of the fine filter in the regulator and the large contaminant particle size required to cause the observed leakage. However, no evidence was found of an excessively high GN_2 dewpoint.

The second, and only other, instance of regulator internal leakage after vendor shipment also occurred during flight acceptance testing of a pressurant control assembly at MMC. After several unsuccessful attempts to flush out any contamination by high flows and cycling, the regulator was removed and returned to the vendor for failure analysis. Upon disassembly, a coating of fine white powder was found on the poppet carrier, on the land area adjacent to the seat, and on the seat itself. The powder on the seat and other internal parts was judged to be Teflon which had separated from Drilube 822 carrier and, in a tacky state, adhered to the regulator surfaces. It is not certain whether the Teflon itself prevented the poppet from seating, or whether the Teflon "caught" a hard contaminant particle which interfered with the seating. The most likely source of the Drilube was in the assembly of the service valves as well as in the test system.

D. Pressure Relief Valve

During normal operation of the MM '71 PSS, pressure reduction from 2516×10^3 N/m^2 (3650 psig) (storage) to 1793×10^3 N/m^2 (260 psig) (utilization) is accomplished by the pressure regulator. The pressure relief valve is installed downstream of the pressure regulator valve and is designed to protect against overpressurization due to regulator failure by venting at 2206×10^3 N/m^2 (320 psig), permitting a nitrogen flow rate of 0.0567 kg/s (0.125 lb/s) without exceeding 2344×10^3 N/m^2 (340 psig) inlet pressure. The outlet of each relief valve is branched to two matched opposing nozzles to minimize any unbalance torques if venting should occur.

The relief valve requirements include:

(1) A burst diaphragm rupture pressure of $2206 \pm 69 \times 10^3$ N/m^2 (320 ±10 psig).

(2) Poppet cracking pressure of 1999×10^3 to 2275×10^3 N/m^2 (290 to 330 psig), reseat at 1999×10^3 to 2206×10^3 N/m^2 (290 to 320 psig).

(3) A flow rate capability of 0.0567 kg/s (0.125 lb/s) N_2 at a maximum inlet pressure of 2344×10^3 N/m^2 (340 psig).

(4) An external leakage rate not greater than 1×10^{-7} STP cm^3/s He at 2068×10^3 N/m^2 (300 psig).

(5) No change in relief pressure setting after 1000 pressure cycles to 95% of relief pressure.

The configuration incorporates two simple mechanical devices: a thin-gage burst disk followed by a spring-loaded sliding poppet. The poppet arrangement provides the capability of repeated on-off operation after burst disk operation so that high pressure is vented only while the level is above the relief valve design value. The burst disk provides an extremely leak-tight seal until an overpressure occurs. A longitudinal cross-section view of the relief valve, shown in Fig. 9, illustrates the burst disk, the main relief poppet, and their respective integrating details. The valve is constructed entirely of 18-8 steel and hard anodized aluminum with a total weight of 1.5 kg (3.3 lb).

The relief valve configuration employed for MM '71 has had extensive qualification and flight confirmation during the Apollo manned space program, being used in both the Service Module and the Lunar Excursion Module. Nonetheless, certain design improvements were required by the MM '71 propulsion subsystem, including the changes required in operating pressures. These improvements and changes are listed below:

(1) Use of an AISI (full hard) stainless-steel burst disk, in place of aluminum, for improved service in corrosive environments.

(2) Use of a welded-in burst disk, to replace an earlier mounting installation for improved leakage service.

(3) Installation of a backup supporting disk on the upstream side of the burst disk to avoid disk damage in the event of reverse pressurization.

(4) Increased length of 0.038 cm (0.015 in.) on the poppet support surfaces to prevent seat chatter during vibrational loads.

The TA and FA tests were accomplished with no problems, as described in Ref. 6. In addition, a margin limit test (MLT) program was conducted as shown in Table 12, with no problems or failures noted.

Further margin limit data was accumulated through the use of two units that were residual from Apollo activity. Minor leaks, less than the minimum acceptable, were noted on the fuel valve, but no leaks were observed on the oxidizer valve, after 6 months' propellant exposure. On the basis of similarity of design, these tests indicated that the MM '71 relief valve was suitable for propellant exposure times that may be experienced during the mission.

E. Pressure Transducers

The high-pressure transducer is a hermetically sealed, potentiometric unit that uses a helical, N_1-Span-C Bourdon tube sensing element to perform over the required pressure range of 0 to 34.5×10^6 N/m^2 (0 to 5000 psia). The feed system pressure transducers were designed, manufactured, and tested by Conrac Corporation, Instrument/Controls Division, Duarte, California. They are designed to operate continuously on the MM '71 spacecraft, 3-Vdc telemetry system. The nitrogen gas to be measured is introduced directly into the Bourdon tube and, therefore, is isolated from the electrical elements and the mechanism. This transducer was an "off-the-shelf" item except that the inlet fitting was modified to meet the brazed propulsion subsystem concept and the case wall thickness was increased to meet the range safety burst pressure requirements. Figure 10 shows the transducer design characteristics. The TA program was conducted according to Ref. 6 with no problems.

The low-pressure transducer is a hermetically sealed, potentiometric unit that uses a convoluted capsule sensing element to perform over a nominal pressure range of 0 to 2758×10^3 N/m^2 (0 to 400 psia). Because this unit is exposed to propellant, the selection of a compatible capsule material that would produce acceptable performance characteristics presented some preproduction problems. The use of the capsule type sensing element of Inconel-X allowed for an all-welded design for long-term propellant exposure.

The free end of the convoluted capsule-type pressure-sensing element is deflected linearly as a function of input pressure. This deflection is mechanically amplified to produce an increased displacement at the potentiometer wiper (Fig. 11). The wiper is designed as a leaf spring to provide uniform contact pressure and is fabricated from a platinum metal alloy. The potentiometer coil consists of platinum alloy wire wound on a cylindrical mandrel. A damping ring is attached to the rim of the capsule to provide a means of vibration damping. The small clearance between the ring and the frame contains a thin film of highly viscous silicone damping fluid.

A major performance problem with the 2758×10^3 N/m^2 (400 psia) Conrac transducer was the shift in output following application of simulated hydraulic shock pulses. During flight operations, the propulsion propellant line pressure transducers are exposed to pressure pulses caused by "water hammer" due to opening or closing of the pyrotechnic valves and engine solenoid valve. Depending on pre-actuation conditions, these pulses can reach peak pressures as high as 13.8×10^6 N/m^2 (2000 psi). The capsule element design proved susceptible to these shocks and it was necessary to reduce the output reproducibility requirements considerably in order to use these transducers without a major developmental program.

During testing of the proof test model (PTM) propulsion subsystem, one 2758×10^3-N/m^2 (400-psia) feed system transducer output became erratic at about midscale. Therefore, when the Flight I propulsion subsystem began leak and proof tests, the outputs of all six pressure transducers were continuously monitored on an analog recorder. Two of the four 2758×10^3-N/m^2 (400-psia) transducers were also noisy. Other assemblies, as well as unmounted transducers, were tested. A total of 23 flight transducers were subjected to noise tests; 17 had noisy output to a varying degree. Noise on seven was sufficient to make the transducers totally useless.

Three transducers, all with extreme noise characteristics were subjected to further tests, including removal of the case for examination of the internal assembly. Two were found to have silicone oil on and around the potentiometric element. The third did not have as much oil around the element, but a considerable amount was visible on the top of the sensing capsule. Also, the wiper arm assembly was loose in the support pivots. No reason could be found why this latter condition would cause noise on the output.

Failure analysis testing suggested strongly that the source of the noise in operation of these transducers was silicone oil coming between the wiper and the coil or potentiometric element. The viscosity of the oil would be high enough to cause the wiper to be lifted off the coil, resulting in an open circuit. In a steady state situation, the preset wiper tension would be sufficient to cause the wiper to sink through the oil and restore continuity with the coil.

Except in the most extreme cases of noise, the signal dropouts occurred only while the input pressure was changing and was in the upper 20% of range. When noise was detected and the input pressure was then held constant, the noise disappeared immediately or within a few seconds. This implies that during conditions of operation at steady state pressures, noise would not be expected to occur. None of the transducers installed on the flight propulsion subsystems have had noise characteristics classified as extreme. Since flight data is taken only during periods of steady state operation, it was decided not to attempt to rework these transducers.

F. Check Valve

The check valve was manufactured by W. O. Leonard, Inc., Pasadena, California, and was used to prevent mixing of the propellants or their vapors in the pressurant manifolds of the MM '71 propulsion subsystem. A cross section of the check valve (Fig. 12) shows the FEP Teflon poppet and the polished seat (AISI Type 321 CRES) against which it seals. The poppet travels in a 321 CRES guide, or "spider," and opens to permit forward flow. The two-piece body is also made from AISI Type 32 CRES and is electron-beam-welded to prevent external leakage. A flow "limiter" ring is used at the poppet/seat interface to force the poppet to stroke evenly at very low flow rates.

The qualification test program was marred by check valve leak problems at the low end of the backpressure requirements (3.4×10^3 to 103.4×10^3 N/m^2)(0.5 to 15 psid). These leaks were caused by metal particles and lubricant residues which were generated by the test fixtures used during acceptance and qualification testing and which accumulated on the valve seat. New procedures required that the fixtures be Freon flushed and thoroughly cleaned before each use and that the lubricant be applied sparingly in the fixtures. After these corrective actions, the check valves passed the qualification testing satisfactorily. However, the excessive leakage due to lubricant and/or particulate contamination did not end at this point but continued as a difficult problem throughout the program following hardware delivery. This problem and two others that occurred during the program are discussed below.

1. Internal leakage. During the period in which the propulsion subsystem and its subassemblies were fabricated and tested, 24 check valves were used, of which 14 exhibited excessive internal leakage; of these 14, nine were disassembled and inspected. Particulate and/or Teflon contamination was found in the sealing area.

The source of Teflon contamination was identified as Drilube 822, a Teflon-impregnated lubricant used by the check valve manufacturer during fabrication and acceptance testing of the unit. For the flight subsystems, the lubricant was changed to Krytox 143 AB grease, which contains no Teflon particles. The source of particulate contamination is not so uniquely identified. The best that can be done is to list, in order of decreasing (qualitative) probability, the sources of particulate contamination to which the unit is exposed:

(1) Particulates generated during the brazing process.

(2) Particulates generated during connection and disconnection of the "B-nut"-type fittings.

(3) Argon, helium, and nitrogen gas supplies.

(4) Clean room environment.

It was concluded that these types of contamination were the cause of all internal leakage failures. Changes to subassembly and subsystem fabrication and test procedures were made in the following manner:

(1) Elimination of brazed-on test fittings at check valve inlet.

(2) Special cleaning of check valve inlet prior to installation on subsystem.

(3) Special in-process leakage tests of check valve to minimize the amount of effort expended on already failed valves.

(4) Elimination of excessive reverse pressure differences applied to check valve.

(5) Elimination of unessential flows through check valve.

(6) Special backup pressure applied during brazing operation.

Following these changes, some improvement in the failure rate was noted, though the sample size was inadequate and the effectiveness of these changes was not fully determined.

It was observed that once subsystem testing had begun with an acceptable check valve leak rate and the fabrication phase had been completed, no check valve internal leak failures would occur. This observation yielded confidence that successful completion of the subsystem assembly phase markedly reduced the risk of a future check valve leak failure. In the future it is recommended that this valve not be used without a "built-in" 10-μm absolute filter at its inlet, and that the use of both Drilube 822 and Krytox 240 AC be discontinued.

2. Flow instability. Instability ("chattering") was noted initially with the preproduction valves. The chattering was recorded with a 700-Hz resonance characteristic. The instability, as ΔP across the valve, ranged from 2.06×10^3 to 142.7×10^3 N/m^2 (0.3 to 20.7 psid) and was produced with flow rates to 3.07×10^3 STP cm^3/s (6.5 STP ft^3/min) GN_2. The amplitude of the pressure oscillations at both the regulator outlet and the point downstream of the check valve (where the relief valve would have been located) varied as a direct function of the regulator inlet pressure. The check valves were tested without the presence of the regulator in order to demonstrate that the valve alone was responsible for the instability. The instability problem was alleviated by a redesign of the flow limiter or flow ring to reduce the clearance between the poppet and the ring. This change did relegate the instability to occurrence at a maximum flow rate of only 1.32×10^3 STP cm^3/s (2.8 STP ft^3/min) GN_2, well below the normal operating point, and reduced the maximum total chatter amplitude to 48.2×10^3 N/m^2 (7.0 psid).

3. Sticking anomaly. During the TA engine firing tests in August 1970, an anomaly occurred with the check valve in the pressurant line leading to the oxidizer tank: the valve appeared to stick or remain closed at the start of flow. During the initial 8-s engine burn, the oxidizer check valve did not crack open normally, and the oxidizer tank operated in a blowdown mode until approximately 6 s after engine start, at which time a less than nominal gas flow rate through the check valve (approximately 0.0014 kg/s, or 0.003 lb/s GN_2) occurred. Subsequently, some special check valve cracking tests were conducted, followed by eight more burns and various pyrotechnic events. During all these events, the check valve appeared nominal.

An investigation involving a number of tests and analyses was conducted to determine the cause of a high cracking pressure in the check valve. Disassembly of the valve revealed the presence of foreign material on the seat, the flow ring, and other surfaces. This material was determined to be forms of nitrate salts, TFE Teflon, and braze material. A dimensional analysis indicated all parts were within drawing tolerances.

In the course of the investigation, some of the theories postulated and investigated were as follows:

(1) Swollen poppet stem interfering with the poppet guide.

(2) Swollen poppet lip interfering with the flow ring.

(3) Contamination between the stem and guide.

(4) Ferric nitrate or other nitrate hardening and adhering between the poppet and seat or between the poppet and flow ring.

Previous information relating Teflon swelling as a function of exposure to N_2O_4 indicated approximately 1% change at 21°C (70°F). Later tests indicated a 2% increase at 49°C (120°F). A series of poppets was exposed to 21°C (70°F) vapor and liquid N_2O_4 and 38°C (100°F) vapor and liquid N_2O_4. Poppet stem diameter and poppet weight were carefully determined over a 2-week interval. The data were plotted and it was determined that 21°C (70°F) full saturation, or complete swell, caused a 1.2% nominal increase in stem diameter. At 38°C (100°F), full saturation caused a 1.8% nominal increase in stem diameter. Data scatter indicated a maximum of 1.3% swell at 21°C (70°F) and 2.1% at 38°C (100°F). Also, previous information indicated that Teflon thermal expansion was 0.08% per 5°C (10°F) temperature change. Two poppet stems were dimensionally inspected at various temperatures. It was concluded that the stem diametral dimension change is a function of both N_2O_4 exposure versus time and thermal expansion of the Teflon. By combining these two effects (using the 0.08% per 5°C (10°F) thermal coefficient), a curve relating stem-to-guide diametral clearance (or interference) as a function of temperature was generated.

A critical conclusion drawn from the results discussed above was that the valve installed on the TA subsystem, S/N 003, may have had an interference fit at 29°C (85°F) and above. The TA valve S/N 003 needed to be saturated at approximately 38°C (100°F) or greater since a cracking pressure of at least $68.9 \times 10^5\ N/m^2$ (10 psid) was observed during the 8-s engine firing tests. This condition is possible, since the propulsion subsystem had been exposed to a 32°C (90°F) to 43°C (110°F) environment a few hours prior to the test.

Based on the N_2O_4 swelling data, the poppet lip will definitely interfere with the flow ring at temperatures above 16°C (60°F). However, by analysis, the resistive force is very small because of the small contact area that exists. A test was conducted to evaluate the resistive force of the lip which interfered with a flow ring, and the force was found to be negligible, confirming the analysis.

The stem from valve S/N 003 was carefully inspected under 30X magnification and no significant deformations in the stem which would be evidence of contamination interference were noted. Also, the stem was gold-plated to amplify imperfections on the surface, and none were found.

Some FEP Teflon material, in contact with 321 stainless steel strips, was exposed to a mixture of N_2O_4 and MMH vapor and allowed to sit for several days. Nitrate material similar to that observed in valve S/N 003 formed on the AISI Type 321 CRES surfaces and the contact points between the Teflon and steel. The material had some adhesion strength, but not enough to support the weight of the Teflon strips.

It was concluded that the TA check valve S/N 003 experienced a high cracking pressure (over 69×10^3 N/m^2, or 10 psid) due to exposure to N_2O_4 vapor at elevated temperatures (over 35°C,or 96°F), resulting in the stem swelling and interfering with the guide. The valve corrected itself through two mechanisms, the reduction in stem diameter due to cooling (to a temperature less than 31°C or 88°F) and probably some mechanical deformation of the stem during movement of the stem in the guide.

On the basis of the testing and analysis, the check valves that were already installed in subsystems were not changed.

G. Filters

The propellant and pressurant filters were manufactured by Vacco Industries of South El Monte, California. Both units employ filtering elements consisting of a stack of metal washer-like disks. Acid-etched grooves, micrometers in depth, on both faces of each disk, constitute the filter passages in the assembled stack. Internal filter details are shown in the sections in Fig. 13. Three problems were encountered with the filters. The first involved excessive pressure drop in the pressurant filters during acceptance testing. The cause was determined to be inadequate thermal control procedures during welding. Repair involved the rebuilding of all the pressurant filters. The second deficiency was excessive contamination in the delivered units. Units were recleaned with improved ultrasonic cleaning procedures to remove this contamination. During this recleaning, rust was found, which was concluded to be a consequence of inadequate processing following etching. Repair entailed the repassivation of the units, in the assembled configuration. The type approval test program for the pressurant filters was completed satisfactorily. Other than the problems previously mentioned, there were no conditions, throughout the program, that indicated failure, deterioration, or marginal performance.

H. Propellant Tank Shell

The Mariner Mars 1971 propellant tank design requirements are described in Table 13. The propellant tank is mounted to the ring structure of the propulsion subsystem by means of four subequatorial tabs. Two methods were used in the development of tank design specifications: (1) conventional pressure vessel stress analysis techniques, wherein standard material property values are used in stress formulas, with arbitrary safety factors applied for the function required and (2) fracture mechanics methods using experimental data on the threshold of crack propagation in the subject material under similar environmental conditions of stress and compatibility.

The propellant tank specifications were initially established around system requirements, propellant compatibility, environmental conditions, and safety factors based on man rating and launch regulations. These specifications stipulated a nominal working pressure of 2068×10^3 N/m^2 (300 psig), a room temperature proof pressure of not less than 3102×10^3 N/m^2 (450 psig) (1-1/2 times working pressure), and a minimum burst of 2.2 times working pressure or 4550×10^3 N/m^2 (660 psig). Continuous pad pressure was limited to 689.5×10^3 N/m^2 (100 psig) $\pm 345 \times 10^3$ N/m^2 (50 psig), and the tank shell was required to withstand complete evacuation.

At the beginning of the MM '71 program, one fracture mechanics data point from an Apollo tank program was available for use. The tank was designed conservatively around the single Apollo data point. The wall thickness was established at 0.078 cm (0.031 in.), later changed to 0.084 cm (0.033 in.), and the tank was heat-treated to a working stress of 496×10^6 N/m^2 (72,000 psi).

Later in the program, more fracture mechanics data were produced by The Boeing Company under contract to JPL. Application of these data to the MM '71 propellant tank disclosed that the basic membrane of the propellant tank ($t = 0.084$ cm or 0.033 in.) was the limiting parameter of the design. All other areas of the tank were determined to be more conservative than the basic membrane. Fracture mechanics analysis based on the experimental data showed that the propellant tanks could be man-rated at a static pressure of 1792×10^3 N/m^2 (260 psi) for N_2O_4/MMH at 29°C (85°F). These allowable pressures satisfied the safety requirement of a total factor of safety of 1.50 on the fracture mechanics design approach, insofar as personnel safety was concerned.

In conformance with the conventional method of tank design, it had originally been planned that three tanks be burst. By these standards, an average of bulk yielding would have been used to establish proof pressure. In this instance, proof pressure (4136×10^3 N/m^2 or 600 psig) was determined from a single fracture mechanics data point and the three burst tanks were primarily for the purpose of giving confidence to this determination. The first tank burst at 5557×10^3 N/m^2 (806 psig). This fact, plus the additional data from the Boeing contract, more than confirmed the fracture mechanics calculations originally made. Consequently, the burst test of the second and third tanks was cancelled.

A 17.8-cm (7-in.) diameter flanged opening was required for insertion of Teflon propellant bladders. The flanged cover for this opening was required to be a minimum weight design and to be a metal-to-metal crush gasket seal for maximum reliability in the face of chemically active propellants. To satisfy these requirements, a titanium flange with a high-density bolt pattern was designed, which incorporated concentric ring serrations sealing against a soft aluminum ring. The outer perimeter of the titanium lip on the tank and the ring serrations at the inner perimeter are driven against a thick aluminum ring. The flange was designed as a flexible plate, and the bolts, lying between the inner and outer perimeters, cause the flange to pivot (toroidal inversion) about its outer perimeter, thereby biting into the aluminum ring at its inner perimeter.

Due to the flexibility of the flange and the levering action described, the joint preload is quite compliant and therefore insensitive to thermal cycling or pressure pulses. In fact, the design is such that when the joint is grossly overpressurized, it leaks severely; then, when pressure is reduced to the design operating range, it reseals to its original helium leak-tight state.

I. Propellant Bladder

The MM '71 propellant expulsion bladder had the initial design constraint that it be either off-the-shelf hardware or be built to off-the-shelf concepts. Selection of a new tank design precluded the first option. It was, however, feasible to build bladders of the size and shape required, using conventional materials and techniques.

The only available bladder material compatible with both the propellants was Teflon. The bladder as it was originally designed was a 74.9-cm (29.5-in.) diameter sphere with uniform 0.025-cm (0.010-in.) walls composed of a laminate of 0.012-cm (0.005-in.) TFE and 0.012-cm (0.005 in.) FEP Teflon. A diametral tolerance of 0.381 cm (0.150 in.) was considered by the bladder manufacturer (Dilectrix Corporation, Farmingdale, New York) to be the minimum range within which work could reasonably be done on a bladder of this size. The bladder had a nominal 2.54 cm × 2.54 cm (1 in. × 1 in.) nipple at one pole and a 17.8-cm (7-in.) diameter × 1.27-cm (1/2-in.) neck fused to a soft aluminum seal ring at the other.

This neck and the seal ring, as shown in Fig. 14, were the only unique features in the initial bladder design. A groove approximately 0.318 cm (1/8 in.) wide by 0.318 cm (1/8 in.) deep was cut into the bonding surface of the aluminum ring. In application, this groove was wrapped full of FEP Teflon prior to the bonding operation. When the aluminum ring was then bonded to the bladder neck, the Teflon in the groove fused with the Teflon of the bladder, forming a secure locking ring. The shear strength of the ring plus the strength of the bond proved to be far stronger than the tensile strength of the bladder material itself.

During the initial subsystem type approval vibration and operational testing, there were several bladder failures. The bladders were from an early prototype run and were of conventional TFE-30/FEP-120 two-layer construction with a uniform nominal wall thickness of 0.025 cm (0.010 in.).

The type of tests conducted and the failures that occurred are as follows:

(1) After exposure to type approval vibration testing using the referee fluids Freon TF and isopropyl alcohol, tears were observed originating in areas of concentric cracks or crazing that appeared near the bladder seal ring, and in some cases around the nipple. The actual ruptures ranged from small holes at the intersection of a crack and crease to massive tears up to 0.61 m (2 ft) long.

(2) After completion of early system hot-firing tests (propellant expulsion) with N_2O_4 and MMH, tears were again observed originating from cracks in crazed areas around the bladder seal ring. The ruptures were generally small, up to 5.08 cm (2 in.) long.

(3) During flight approval and type approval levels of low frequency vibration testing using water as the test fluid in a Plexiglas tank, major tears occurred, originating at a point near the seal ring on the axis of vibration and extending up to 2 ft into the bladder body. No areas of crazing were apparent.

(4) During tank filling operations, tears developed in the nipple area because of excessive bladder ΔP without proper tank backup.

The testing was accomplished with conventional gas ullage orientation, i.e., the pressurized gas ullage was located on the outside of the bladder (Fig. 15). Slow motion photography of the Plexiglas tank tests showed that during vibration, at each point of load reversal, the liquid would slosh against one side of the tank, and an accompanying snapping or whiplash type of motion would occur in the bladder neck at the opposite side of the tank. The bladder material could not withstand the resulting stress and a tear would develop in the neck area.

After evaluating the test history, it was apparent that the conventional bladders, even with thickened necks, would not withstand the launch environment that was being imposed upon them, particularly with the obvious deterioration resulting from contact with referee fluids and propellants. Analysis revealed that the failures originated from areas of surface cracking that occurred in the outer FEP Teflon layer. These cracks were determined to be "solvent stress cracks," which can result from exposure of low-molecular-weight FEP, such as Type T-120, in contact with propellants or referee propellants. To solve the solvent stress cracking problem, it was proposed that the more recently developed TE-9511-type FEP be substituted for the T-120 FEP. This new type of FEP is less sensitive to solvents and has a higher molecular weight than the T-120 FEP. Because of the implications and seriousness of the bladder failures being experienced, a major effort to evaluate the new material and structure was initiated.

TFE/FEP codispersion film samples representative of bladder material were prepared and submitted to test. Significant results of these tests are summarized in Table 14 (see Ref. 7). As-fabricated test specimens are identified as "dry"; specimens that were soaked in the various liquids are identified as "wet." Heptane exposure was included in the test program as a reference, because heptane is a known degrader of FEP Teflon. The test results clearly show the advantages of the new codispersion film, compared with the TFE/FEP T-120 laminate material. The TFE/FEP TE-9511 codispersion samples did not show evidence of "solvent stress cracking" which had been experienced on the TFE/FEP T-120 laminate bladders.

As a result of these and other tests, new bladders of the new material were procured. With the exception of minor design changes, the new bladders were geometrically identical to the earlier ones (a spherical shaped Teflon membrane bonded to a soft aluminum seal ring).

Figure 16 shows a cross section of the TFE/FEP codispersion membrane used for the new bladders. The outer and inner layers were formed from a codispersion consisting of 20% TE-9511 FEP mixed with 80% of TFE (T-30) (percentages are by weight). The inner permeation barrier consisted of 100% TE-9511 FEP. These bladders were of uniform thickness throughout, as opposed to the original TFE/FEP laminate design, which incorporated a thickened neck and nipple areas. The two significant revisions to propulsion subsystem operational procedures that influenced membrane design were also instituted:

(1) Gas ullage relocation. The initial launch phase propellant tank gas ullage was relocated from the gas to the liquid side of the bladder (Fig. 15). This allowed the entire bladder to be in contact with the tank inner wall and to be in an unstrained condition during the launch vibration mode, as opposed to having the neck area in a highly strained condition with the ullage on the gas side of the bladder. As a result, it was not necessary to thicken the neck area.

(2) Maximum differential pressure. A maximum differential pressure across the membrane (liquid to gas side) during all prelaunch operations was established at 206.8×10^3 N/m^2 (30 psi) for a wet bladder and 275.8×10^3 N/m^2 (40 psi) for a dry bladder. This, in combination with closer dimensional control between the bladder nipple and the tank nipple receptacle precluded the necessity for a thickened nipple area.

An extensive test program was undertaken to qualify the new bladders. Several tanks with new bladders were subjected to subsystem vibration tests in three-axis vibration at type approval level. Following vibration, each tank assembly was subjected to propellant explusion tests at 0°C (30°F) and 32°C (90°F) propellant temperature.

In addition, low-frequency vibration testing (slosh) was conducted with propellants and referee fluids to qualify the tank assembly for transportation and launch vibrations (Ref. 8). The general scope of the bladder penalty tests required that the bladders withstand flight acceptance and type approval subsystem level vibration with solvents, followed by a vacuum dry and bladder leak check, TA subsystem vibration, and low-frequency, single-tank vibration with propellants, and demonstrate at least a 96.5% expulsion efficiency. In addition to these tests, three extra bladder cycles were included, using temperature-conditioned propellants. All these tests were successfully passed and the flight subsystems retrofitted with the new bladders just prior to shipment to the Air Force Eastern Test Range.

J. Propellant Tank Standpipe

The purpose of the MM '71 propellant standpipe assembly is to prevent pressurant gas, which may be located within the propellant tank positive expulsion bladder, from discharging with the propellant to the rocket engine. Each propellant tank and bladder set for the MM '71 propulsion subsystem (i.e., oxidizer and fuel) has its own standpipe, which collectively constitutes the propellant tank assembly. Without the use of a gas retention device, such as the standpipe assembly, gas pockets intermingling with the propellant could be injected into the engine and could cause erratic engine performance.

The standpipe assembly consists of (1) screens which prevent gas within the propellant tank bladders from being drawn to the engine and (2) a trap which allows the propellant lines to remain full of liquid in the prelaunch and launch "engine-up" configuration. Figure 17 illustrates the assembly as designed in concept by the Rocketdyne Division of North American Rockwell Corporation, Canoga Park, California, under JPL contract, and integrated into the tank. The fabrication of the standpipe was performed by Pressure Systems, Incorporated, Los Angeles, California. The proper function of the standpipe depends on the bubble retention properties of a primary and a secondary screen element (0.076-cm (0.030-in.) diameter and 0.051-cm (0.020-in.) diameter holes respectively) and the quasi-enclosed configuration of the secondary trap. The isolated volume of the secondary trap provides for storing gas-free propellant for the first engine burn and for the entrapment of any subsequent gas flow during succeeding firings. Normal liquid flow proceeds across the perforated cone, through the primary screen to the riser element, around to the baffled inlet section of the secondary compartment, across the secondary screen, and onward to the tank outlet. The inlet section to the secondary compartment is purposely baffled with horizontal and vertical vanes to prevent gases from entering the secondary trap under the influence of launch sloshing disturbance. The inlet section also allows for temperature-induced liquid/gas surface excursion, during no-flow modes, without exposing the secondary trap entrances to gases. The requirements and guidelines for the propellant standpipe acquisition trap are shown in Table 15.

An extensive test program was conducted to characterize and evaluate the propellant acquisition trap according to Table 16. Functional testing of the standpipe at 1 g proved to be difficult, but the assembly had no history of failure.

K. Feedline Hose Assembly

The MM '71 feedline hose assembly consists of an extruded tetrafluoroethylene resin (Teflon) tube, externally reinforced with a single-layer, corrosion-resistant steel braid. The assembly incorporates permanently attached (swaged) end fittings at each end. The end fittings are of the 37-deg flared tube design, employing permanently attached connection nuts. The pertinent physical dimensions of the hose assembly are as follows:

	cm	in.
Length, overall, flow path	53.3	21
Length, flexible portion	42.5	16.75
Internal diameter, flexible portion	1.031	0.406
Wall thickness, Teflon liner	0.109	0.043
Outside diameter, flexible portion, over braid	1.427	0.562

Two identical hoses, one fuel and one oxidizer, are employed to convey propellants from the rigid propellant supply system to the movable rocket engine assembly. As shown in Fig. 18, the hose assembly is permanently preformed into a "question mark" configuration to permit installation within the available spatial envelope and to minimize hose resistance to engine motion imparted by the gimbal actuators. The hose assembly design was specified to utilize a nonelectrically conductive Teflon liner because previous experience with conductive-lined hoses in similar applications had shown a tendency for the conductive hose lining material to become detached from the hose interior and contaminate downstream regions of the system.

A program of type approval testing, shown in Table 17, was performed to determine that the proposed hose design was adequate to satisfactorily meet the requirements imposed by the operational and environmental conditions to which it would be subjected during the MM '71 flight mission.

Two problems were associated with the feedline hose assembly. The first was the failure of feedline hose assemblies by discharge of flow-induced electrostatic charges.

A series of tests was performed to characterize and define an observed tendency for the feedline hose assembly to develop pinholes in the Teflon hose liner. These formed as a result of the discharge of flow-induced electrostatic charges from the hose liner inner surface, through the hose liner wall, to the electrically grounded exterior metal braid. Discussions with the hose manufacturer and review of the available literature dealing with hose failures of this nature indicated that the source of the electrostatic voltage required to cause the observed failures was the hose cleaning process employed immediately prior to the detection of the failures.

The cleaning process consisted of flowing filtered Freon Precision Cleaning Agent through the hose at approximately 0.126 liters/s (2 gal/min), while the exterior metal braid of the hose assembly was electrically grounded. Failure of the hose liner occurred in three out of four hoses cleaned in this manner. Comparison of the cleaning process with the experimentation described in the literature established a high degree of similarity in the parameters required to generate a high electrostatic voltage level on the hose interior. These items of similarity are:

(1) Low electrical conductivity of the hose liner.

(2) Presence of cellulose-type (paper) filter immediately upstream from the hose.

(3) Low electrical conductivity of the flowing fluid.

A test program was devised to characterize the electrostatic charge generation phenomenon with regard to (1) flight operation of the MM '71 propulsion subsystem, and (2) cleaning of hose assemblies.

The following conclusions were reached as a result of the test program:

(1) The feedline hose assemblies are suitable for use on the MM '71 propulsion subsystem. The highest voltage observed during propellant testing was 5000 V; this is less than 1/10 of the minimum observed hose failure voltage and approximately 1/8 the minimum theoretical breakdown voltage for 0.101-cm (0.040-in.) wall thickness Teflon tubing reported by other experimenters. Voltages generated in the fuel hose are so low that they can only be described as negligible.

(2) Extreme care must be exercised in cleaning nonconductive Teflon hoses in order to avoid pinholes caused by electrostatic discharge. The problem can be minimized if cleaning fluids with a high electrical conductivity can be utilized, if flow rates of cleaning fluids can be kept low, if upstream filter assemblies are kept as far from the hose assembly as possible, and, most important, if the hose exterior is not grounded electrically with respect to the cleaning system.

The second problem with the feedline hose assembly was associated with the quality of end fittings. The hose assemblies were originally procured with standard end fittings according to MIL-F-27272A. The nuts utilized for those fittings were machined from extruded hexagon bar stock, Type 304 CRES. The extruded hexagon surface forms the external wrenching surface of the nut and is not further machined during nut manufacture. The extrusion process leaves a surface condition of low quality; typical specimens exhibit pits, cracks, and generally poor finish caused by inclusion of oxides, die lubricants, and other impurities during forming.

During dye penetrant inspection of the hose end fittings, all nuts were found to be of low surface quality, and three were found to have distinct cracks or linear oxide inclusions. Because improper function of the hose assembly through nut failure would affect MM '71 flight success adversely, and because the hose environment contained the essentials for the occurrence of stress corrosion cracking, the decision was made to remove and replace all hose nuts with new ones made from Type 304 vacuum arc remelt steel.

L. Service Valves

The service (fill) valve is a manually operated, threaded stem, ball-and-seat closure design. It is a compact, light-weight, essentially zero-leakage valve and has exhibited high reliability on several flight programs including Mariners 1964, 1967, and 1969. The valve is used at points of system access for propellant fill and drain, gas pressurization, and system bleed. The operating assembly consists of a flight half permanently attached to the spacecraft (Fig. 19) and a ground half that is attached to lines from the ground servicing equipment (Fig. 20). A separate cover, incorporating a redundant seal, encloses the open end of the flight half when it is disconnected from the ground half. Service valves were manufactured by W. O. Leonard, Inc., Pasadena, California.

The operating mechanism of the valve is a hard, smooth-finish, self-centering ball, loosely caged at the end of the threaded stem, which seats itself on the edge of a hole drilled into the soft metal of the valve body. The ground half, used to open and close the airborne half, consists of an adapter nut, drive shaft with wrench flats, AN fitting, ball bearings in a retainer to support the shaft, and the necessary Teflon omniseals.

This unit was used with hydrazine on the MM '69 mission and was modified for bipropellant use on MM '71. The major modifications were:

(1) Replacement of rubber O-rings with Teflon omniseals.

(2) Substitution of a compatible lubricant.

(3) Replacement of noncompatible cadmium-plated bearings with stainless steel bearings.

No problems occurred during FA and TA testing of these valves. During subsystem testing, however, there was a high rate of service valve internal leakage failures. The cause of these failures was concluded to be twofold:

(1) Inadequately lapped valve seats and insufficiently smooth ceramic balls, which resulted in internal leakage failures. This problem was corrected by replacing the ceramic balls with ones of higher quality and by replacing the valve seats.

(2) Excessive valve and valve cap closure torques, resulting in seat loads sufficiently large that the stainless steel (304L) seat material work-hardened, became brittle, and subsequently flaked off or spalled. This was corrected by reducing the closure torques to

(a) Valve -- 169 cm-N (15 in-lb_f)

(b) Valve cap -- finger tight and backed off one hexagon flat for all but the final closure operation, where the closure torques were 282 cm-N (25 in-lb_f) for both valve and valve cap.

V. ROCKET ENGINE ASSEMBLY

A. Description

The MM '71 engine, shown in Fig. 21, is a two-piece conductively cooled combustion chamber and radiation-cooled nozzle extension weighing 7.7 kg (17 lb). The combustion chamber, fabricated from hot-pressed beryllium, is attached to the 40:1 cobalt alloy nozzle extension by a Rene 41 nut. The aluminum alloy injector assembly consists of 36 unlike doublet (oxidizer on fuel) elements and 48 boundary layer coolant (BLC) orifices as shown in Fig. 22. This engine employs an axially located acoustic cavity for combustion stabilization. The interface between the injector and combustion chamber and associated hot gas seals can also be seen in this figure. The engine employs a unique method of thermal control developed by the Rocketdyne Division of North American Rockwell Corporation and termed "INTEREGEN." A schematic of the conductive cooling process showing the heat flow route can be found in Fig. 23. Heat transferred convectively to the engine is conducted through the thick, highly conductive chamber walls and transferred, again convectively, to the BLC covering the thrust chamber walls near the injector. The BLC covering is also convectively heated from the hot gas side. In this manner the engine can run indefinitely with steady temperature distribution. Success of this cooling technique depends on the heat absorption capabilities of the BLC and the proper thermal management in the metal walls so that adequate protection from the hot combustion gases is afforded.

The MM '71 engine is a modification of the North American Rockwell Corporation (Rocketdyne Divison) Minuteman III Post Boost Propulsion System axial engine. This 1406-N (316-lb_f) thrust engine, designated as the RS1401, does not operate in the steady-temperature mode but rather operates in a quasi-heat-sink mode with chamber temperatures continually rising during a relatively short burn time. Modifications to this design were incorporated which permit the steady state mode and, hence, support the more demanding MM '71 duty cycle; i.e., the continuous Mars orbit insertion burn of up to 1000 s duration. A 40:1 expansion ratio 80% bell Haynes 25 alloy metal nozzle was designed for optimum vacuum operating conditions and was incorporated in the engine design utilizing a joint between the beryllium thrust chamber and the nozzle downstream from the throat at the 3.3 to 1 expansion ratio point. Figure 24 shows the MM '71 engine.

The MM '71 engine is equipped with a torque-motor-operated, mechanically linked bipropellant control valve produced by the Moog Corporation, Aerospace Division, East Aurora, New York. During the mission the valve must contain propellants or their vapors for up to 200 days without leakage and operate within specification limits after exposure to a temperature and pressure profile that fluctuates with mission events. It was necessary, therefore, to determine the effect of long-term, high-temperature storage upon the sealing and operating characteristics of this valve. A photograph of the valve mounted on the engine is presented in Fig. 25.

Three valves were entered in the storage program. Two of these valves were stored at upper limit (margin limit) of pressures and temperatures expected in the Mariner mission. The third valve was stored at nominal mission pressure and temperature conditions. Test results were positive, indicating that the valve could meet the long mission duration requirements.

The engine operating conditions are listed in Table 18 and engine requirements are listed in Table 19.

B. Development

1. Prototype test program. The purpose of this program was to test the MM '71 modification to the RS1401 for acceptability in meeting more stringent spacecraft requirements. The RS1401 had been previously qualified for the Air Force application.

The test program, consisting of fabricating two engines to the selected prototype configuration, was then conducted to determine operating characteristics. Fourteen vacuum tests were conducted with engine S/N 4098603, accumulating a total firing duration of 4245 s. Twenty-one vacuum tests were conducted with engine S/N 4098604 accumulating a total firing duration of 4620 s. A nominal vacuum specific impulse of 2824 N-s/kg (288 $lb\text{-}s/lb_m$) at 1.55 mixture ratio and 8.06×10^5 N/m^2 (117 psia) chamber pressure was established for both engines. Four steady state tests of 840 s duration were accumulated on each engine with two of each set with GN_2 saturated propellants. No instabilities were encountered at the nominal run conditions. Six reduced thrust tests with engine S/N 4098604 indicate unstable operation below 445 N (100 lb_f) thrust with unsaturated propellants and below 890 N (200-lb_f) thrust with GN_2 saturated propellants. Reduced thrust levels were achieved by reducing run tank pressures. The results of the test program (fully described in Ref. 9) indicated the capability of this engine to meet the MM '71 performance and duration requirements.

A simulated sinusoidal launch vibration test was conducted with a prototype engine. Since the MM '71 vibration environment was higher than the RS1401 vibration environment, there was concern regarding the engine structure. This concern proved justified as a high response resonance of the engine gimbal ring assembly occurred at approximately 95 Hz and resulted in high-g forces which damaged the gimbal bearings and exceeded design limits of the propellant valve. Information derived from these tests was utilized for a redesign of the prequalification engine gimbal ring assembly.

2. Margin limit test program. A series of tests was performed with a prototype engine at Edwards Test Station. This engine had previously been extensively tested by the engine contractor. The test program was intended to define the operating limits of the engine, in particular the maximum values of mixture ratio and chamber pressure that would permit acceptable INTEREGEN cooling. Post-firing thermal vacuum soak data was to be obtained, since facility limitations prevented Rocketdyne from obtaining such data. Much of the information (performance, thermal characterization, etc.) obtained in the program complemented that acquired by Rocketdyne in their prequalification and qualification testing of the basic engine.

The tests, which encompassed firing durations from 16 to 900 s, operation with propellant temperatures over the range of 4 to 34°C (40 to 93°F), and GN_2 saturation levels from 0 to 100%, were followed by thermal soak periods in vacuum for as long as 3 h. The engine was operated over a mixture ratio (O/F) range from 1.29 to 1.75 and a chamber pressure range from 758 to 889 N/m^2 (110 to 129 psia). The shift in mixture ratio caused by operation with GN_2 saturated propellants over a range of temperatures was determined. Start and shutdown transient time intervals, using a flight-simulated Moog valve driver circuit, were determined for all tests. Significant results are as follows:

(1) Start and shutdown transient times recorded in the test series, wherein "time-zero" is defined as the application of voltage to the coils of the Moog valve, are as follows:

Δt_1 -- Signal to Moog valve to open = 27 ms

Δt_2 -- Signal to positive P_c rise = 43 ms

Δt_3 -- Signal to 90% full P_c = 100 ms

Δt_4 -- Signal to Moog valve to close = 9 ms

(2) It was concluded from the test data that a nonsaturated operating O/F of 1.67 represents the highest safe O/F for the engine, based upon INTEREGEN cooling limitations. This provides a 0.10 margin over the nominal mixture ratio.

(3) No O/F shift takes place at saturation pressures below about 1034 N/m^2 (150 psia), which corresponds to about 60% of the saturation level at the nominal MM '71 operating pressure of 1758 N/m^2 (255 psia). Above this saturation level, the O/F shift is a strong function of propellant temperature, ranging from about -0.07 units for fully saturated 21°C (70°F) propellants to as high as -0.17 for fully saturated 34°C (93°F) propellants.

(4) Based upon the data from a single 900-s firing performed at two different P_c levels, 848 N/m^2 (123.0) and 886 N/m^2 (128.5 psia), it is concluded that the maximum chamber pressure for acceptable INTEREGEN cooling is about 848 N/m^2 (123 psia).

One of the requirements imposed on the engine is that its roll or swirl torque be no more than 16.9 cm-N (1.5 in.-lb). It was necessary to demonstrate by firing tests that the roll torque induced in the MM '71 engine would be within specification limits so that it could be counteracted by the cold gas attitude control system. Based upon data from four engine firings in a special test stand, the measured torque ranged from 1.13 to 11.3 cm-N (0.1 to 1.0 in.-lb_f), with a measurement uncertainty of about ±4.5 cm-N (±0.4 in.-lb_f).

C. Prequalification Engine Design and Fabrication

During the prototype program, the MM '71 engine generally met the requirements of the MM '71 Mission. However, problems encountered included (1) a low-frequency resonance during vibration, (2) gas leakage in the joint between the nozzle extension and the thrust chamber, following hot-fire tests, and GN_2 saturation of the propellants causing shifts in engine mixture ratio calibration during hot-fire tests. In order to better qualify the MM '71 engine for the MM '71 requirements, several design changes were made and incorporated in the prequalification engine. A test program was then conducted to determine the worth of these changes as well as to gain additional data regarding the effects of propellant saturation on engine operation.

The most serious problem encountered in the prototype test program was the failure of the engine gimbal ring design to withstand the launch vibration requirements of the MM '71 mission. This gimbal ring was redesigned with a larger square cross section, and preliminary vibration tests were conducted, with a mockup engine, to verify the design. Although the new gimbal ring itself was satisfactory, the bolts attaching the bearing housings to the gimbal ring became loose during random vibration. This was corrected by changing the threads to allow greater torque to be applied and by adding a lockwire and "loc-tite" adhesive to further prevent loosening of the bolts. This design was incorporated in the prequalification engines used in the test program.

During the prototype program, it was discovered that the joint between the nozzle and the thrust chamber was not leak-tight after hot firing. No evidence of hot gas leakage could be found by visual inspection of the joint parts. However, it was decided to change the joint design slightly to attempt to reduce the leakage. The changes included a change from Acme "V" threads on the joint nut to buttress-type threads, increasing the holding finger preload by tightening the nut further, and improving the match between the thrust chamber and nozzle mating surfaces by changing tolerances and dimensions. These changes were incorporated in the prequalification engines.

D. Prequalification Engine Test Program

During the prototype hot fire testing, a shift in mixture ratio occurred when the tests were conducted with propellants saturated with GN_2. This shift was approximately 0.1 O/F lower, primarily because the GN_2 in solution in the oxidizer was released in the lower pressure injector manifolds. This release of gas increased the oxidizer side effective volumetric flow rate, thereby decreasing oxidizer flow and reducing the mixture ratio. A similar effect occurred on the fuel side, but because of lower GN_2 solubility in fuel the change in fuel flow was much smaller and was difficult to separate from normal operating variations.

A series of hot fire engine performance survey tests was conducted during the prequalification program using propellants at various temperatures, saturated with GN_2, as well as propellants at various temperatures with no GN_2 in solution.

The engine prequalification test program, shown in Table 20, was conducted at Rocketdyne. The results of this test program are presented in Ref. 10. Two engines, P.N. RS000601 (S/N 0003 and S/N 0004), started through the test program; however, engine S/N 0004 was reassigned to the TA test program after completion of TA-level vibration testing, and engine S/N 0003 went on to complete the prequalification test series. Table 21 summarizes the hot fire test portion of that program.

E. Type Approval Engine Test Program

The MM '71 TA test program was conducted at Rocketdyne on the Mariner '71 rocket engine assembly, S/N 0004 and S/N 0005. The type approval test program included both temperature-humidity and launch vibration tests in addition to extensive altitude hot-fire testing. Included in the hot-firing testing were mission life cycle and performance survey tests, conducted under various environmental conditions providing data for parametric characterization of the engine. Reference 11 is a detailed report of the type approval test program.

The purpose of the TA test program was to demonstrate that the MM '71 engine exceeded the requirements of the mission and was suitable for flight. A secondary objective was to obtain a parametric characterization of engine performance by exposing the engines to the extremes of environmental operating conditions defined in the JPL engine model specification (JPL Spec. SS 504869).

The test program sequence is listed in Table 22 and the test matrix is shown in Table 23. Prior to being placed in the TA test program, both engines completed flight acceptance testing. The flight acceptance sequence is shown on Fig. 26.

After completion of the environmental tests, the engines underwent an extensive hot-fire test series. During the TA series, the engines accumulated a total of 6,307 s during 23 starts. Each engine completed the equivalent firings of more than three missions.

During the type approval Program, three significant problems occurred, involving the chamber pressure transducer, leakage of the engine primary "V" Seal, and plugged or obstructed injector orifices. The pressure transducer problems are discussed in Section V-F. The other two problems are discussed below.

Both TA engines were subjected to leak and functional tests following the completion of the first mission duty cycle hot-fire test sequence. Engine S/N 0004 was found to leak 2.5 STP cm^3/s nitrogen gas past the injector-to-thrust-chamber gold-plated Inconel "V" Seal, and engine S/N 0005 was found to have a "V" Seal leakage of 1.67 STP cm^3/s. No leakage was allowed by specifications. Both engines continued on through the remainder of the TA program and successfully completed a total of three mission duty cycles.

Leakage past the primary "V" seal occurred after completion of a mission duty cycle due to fracture of a hard, brittle intermetallic phase which formed at the gold-beryllium interface. The presence of the hard phase was due to diffusion of beryllium into the gold plate during hot-fire testing.

Engine S/N 0004 had completed the first mission duty cycle when a slight wrinkle in the nozzle extension was discovered. This problem had not shown up during the FA hot-fire acceptance testing. It was decided to continue the TA hot-fire program, but the planned firing sequence was changed to delete the minimum-impulse short-duration pulses and to conduct the mixture ratio survey last because of the possibility of overheating this area during margin limit operating conditions. In addition, the nozzle exit thermocouple was relocated to the deformed area to monitor temperature conditions at this hot spot. The engine successfully completed the TA program with no other problems.

The buckling of the nozzle extension was caused by a hot streak in the combustion pattern. This in turn, was caused by plugged fuel orifices allowing oxidizer streams to impinge directly on the wall. The fuel orifices were plugged by droplets formed when the boundary layer cooling manifold ring weld beam broke through into the fuel manifold.

The electron-beam current level of 9.0 mA used on the weld of the discrepant injectors yields excessive penetration. When the penetration level is near the depth of the material being welded, the normal ±5% spiking inherent in the welding process causes the beam to break through intermittently, forming the metal droplets seen in these injectors. The reduction in weld current to 8.5 mA on the flight engines was adequate to prevent breakthrough.

The engine injector orifice flow sensor, a hot-wire anemometer flow measuring device, was developed for use on the MM '71 program after the plugged injector problem occurred. Two flow sensor probe units and one test control console were fabricated. A flow record for each flight injector was maintained throughout the buildup and testing of each engine at the propulsion subsystem level and at the spacecraft level. This made it possible to know the condition of all injector orifices prior to spacecraft launch at AFETR. Any significant changes or anomalies that developed during spacecraft buildup and testing were recorded and were the subject of engineering review. Specifically, the status of the injector orifices was determined before and after propulsion subsystem FA vibration and FA vibration with the spacecraft, and just before propellant loading at AFETR. No problems of injector blockage occurred, other than those associated with high weld current.

F. Engine Chamber Pressure Transducer

The chamber pressure transducer used on the MM '71 1334-N (300 lb_f) thrust rocket engine is a potentiometric type having a range of 0 to 1379 X 10^3 N/m^2 (0 to 200 psia). The element resistance (5000 ohms) utilized an excitation voltage of 3 V dc. The operating temperature range is -7 to 104°C (20 to 220°F). The units were built and acceptance tested by Servonics Inc., a Division of Gluton Industries, Costa Mesa, California, prior to assembly to the engine by the engine contractor.

During the initial engine hot-fire TA test sequence, at approximately 155 s into the 860-s test, the chamber pressure output became erratic, and the test was finally terminated at 250 s because chamber pressure output was reading below the minimum pressure of 690 X 10^3 N/m^2 (100 psia).

Subsequent failure investigation revealed that the intermittent electrical signal was caused by the buildup of a high-silicon-content deposit on the operating pressure area of the coil. The source for this deposit material was not established definitely but there were several possible candidates. Since the source of this deposit was not determined, no plan for elimination of the deposit was established. All flight transducers were subjected to a 900-s hot-fire test using a test engine. No problems were encountered during this test program. Therefore, it was concluded that the problem transducer was an isolated case.

VI. PROPULSION SUBSYSTEM ASSEMBLY

Early in the design phase of the propulsion subsystem, the comparisons and tradeoffs between welding and brazing of tube-to-tube and tube-to-component joints resolved themselves to a choice of the induction brazing process.

A. Brazing Process Development

Aeroquip equipment (Aeroquip Corporation/Aircraft Division, Jackson, Michigan) was utilized, consisting of (1) a 15-kV water-cooled induction generator/voltage regulator combination, (2) a remote console, which was connected via RF cable, water cooling, and argon gas lines to the induction generator and (3) the water-cooled braze tools. A considerable number of qualification brazes of each fitting size and configuration were made during the course of the training program and, based upon this experience, the buildup of the propulsion dynamic test model and the first flight prototype propulsion subsystem (engineering test model) was initiated.

During the course of the braze development and early stages of the assembly buildup, special considerations relative to cleanliness and preparation of material and techniques for maintaining inert environments in the braze joint zone were established. The braze fittings of 304L stainless steel with 82%-18% gold-nickel alloy were procured from Aeroquip to a specification requiring cleanliness to a 25-µm maximum particulate level. Tubing and components to be joined were processed in a specific sequence to assure proper preparation for brazing and the maintenance of cleanliness. Tubing to be used for interconnecting subassemblies was procured on a special purchase to assure the close control required on the outside diameter. After bending, the tube assemblies were rough-cleaned to remove oils, greases, and salts in an alkaline soak process (Turco 4215-6 water solution). The rough-cleaned tubing was then sized by temporary installation on the subsystem. Braze fitting simulators (fixtures sized to represent the internal configuration and length of the braze fitting) were used to establish the proper fit of the interconnecting tubing. The sized tubing was prepared by the mechanical abrasion and chemical etch process in the braze region and then detail-cleaned and packaged in preparation for final assembly.

The subsystem assembly procedure and the braze procedure listed detailed steps to be followed to eliminate moisture and oxygen from the braze region. A hygrometer sensing probe was installed on the system to be brazed to measure the argon effluent for moisture content. A specification of less than -57°C (-70°F) dew point was required prior to committing to the braze cycles. An example of these purging and blanketing requirements was the brazing of tubing connections to the pressurant tank assemblies. Since this assembly terminates in a significant dead-end volume (the tanks themselves), the technique employed was to evacuate the system using a vacuum pump and backfill with argon gas. This removed the air and moisture from the system. The argon was then vented to atmospheric pressure to assure that hot braze would not be blown out during the brazing process. The braze was then completed in a normal manner.

Another example of changes in the braze procedure was the preparation for the braze of the extremely contamination-sensitive check valves in the pressurant check and relief valve assembly. Early experience in brazing the inlet manifolds to the check valves was poor, in that contamination resulting from the braze sequence was allowed to enter the check valve poppet/seat area and cause leakage. Procedures were changed to eliminate, as much as possible, any flow through the check valves during the brazing cycle. This was accomplished by back-pressurizing the check valves against the inert gas purge pressure. This technique improved but did not eliminate the leakage failures of check valves. Replacement of check valves was required on two of the flight subsystems. During all brazing cycles, an argon inert gas-blanket emanated from the braze tool and surrounded the braze fitting to preclude oxidization due to inflow of atmospheric air.

The repair procedural sequence requires the cutting through of the faulty braze fitting, removal of the remaining stubs, repreparation of the tubing or component braze region, and rebrazing with a new braze fitting. The need to minimize the level of contamination generated by the rework sequences and the specific MM '71 equipment requirements necessitated redesign of some of the Aeroquip repair equipment. One important change was in the scarfing (or cleaning) tool used to remove excess braze material and to resize the tubing ends following stub removal to accept a new braze fitting. The original design utilized an O-ringed pilot inserted into the tube to guide the cutter of the cleaning tool. Lack of positive concentric cutting and the contamination generated during insertion of the pilot led to the need for a new tool. This new tool clamped to the outside of the tube and used an expandable plug to eliminate cuttings from the interior of the tubing. An inexpensive "throw-away" design was also made to replace the stub removal tool.

B. Subsystem Assembly Experience

The first subsystem to be assembled was the dynamic test model, which used a flight-like structure (truss assembly, propellant tank ring frame, and thrust plate). The subassemblies, with the exception of the propellant tanks, pressurant tanks, and engine, were mass mockups. Although the primary function of this subsystem was to provide a close simulation of mass distributions of the propulsion system for spacecraft dynamic and static load testing, it also provided a stepping stone for the generation of the operational subsystem buildup procedure.

The ETM was the initial subsystem to be assembled with fully operational components and subassemblies. The assembly of this system provided the necessary experience to be used in buildup of the type approval model and the three flight subsystems. The ETM was used to establish the tubing runs for interconnection between subassemblies. From the as-built configuration, the tube assembly design drawings were made. These drawings were then used as the basis for tube fabrication for all succeeding subsystems. Since the function of the ETM was to characterize the operation of the flight subsystems under all anticipated modes, as much instrumentation as possible was incorporated without disturbing the hydraulic characteristics of the subsystem. Flowmeters were added to the inlet lines of the propellant isolation assemblies; thermocouples were installed at several points to assess the thermal behavior under firing conditions; and strain gauges were mounted on the subsystem to determine and to control the input levels of induced vibration during environmental testing. In addition, the pressurizing lines leading to the propellant tanks were prepared with AN and Swagelok fittings to facilitate removal and replacement of the tanks.

The type approval model was assembled using sequences and equipment identical to that for the flight subsystems. Some deviations were required, but in general this subsystem was assembled in a fashion representative of the flight subsystems.

The assembly of the proof test model was accomplished in the same sequence as that for the flight subsystems; however, following spacecraft testing, a complete disassembly and reassembly of this subsystem was required. Replacement of some components and rework of the structure were required as a result of redesign dictated by failures during test.

The experience gained throughout the assembly phases of the preceding subsystems, as well as necessary design changes dictated by problems during the testing phases of each subsystem, was used during the assembly of the Flight I and II subsystems. The important problems which resulted in changes to the assembly procedure are discussed in the next section on flight subsystem testing.

VII. SUBSYSTEM FLIGHT ACCEPTANCE TESTING AND FLIGHT PREPARATION

At the completion of fabrication, each flight propulsion subsystem was subjected to the following test sequences:

(1) Proof and leak

(2) Functional

(3) Vibration

(4) Vacuum chamber leak

(5) Post-vibration functional

A. Proof and Leak Test

The purpose of the proof test portion of the proof and leak test was to demonstrate integrity of the subsystem at levels of pressure 1.5 times the normal working pressure for various components of the subsystem. The levels of pressure for parts of the subsystem varied from 41.4×10^6 N/m^2 (6000 psi) for the pressurant bottles to 1723×10^3 N/m^2 (250 psi) for the rocket engine. Certain areas of the propulsion subsystem could not be proof-pressure tested. These included the burst disk of the oxidizer and fuel relief valves and that portion of the tubing between the pneumatic regulator and the check valves. Since the regulator locks up at slightly above flight pressure there was no way of increasing the pressure downstream of the regulator and between the check valves to 1.5 times the working pressure. Table 24 contains the proof and leak test pressures for the subsystem.

The purpose of the leak test was to verify that zero leakage was obtained at the many braze joints of the subsystem which had been added to interconnect subassemblies. Helium gas was used as the leak detection medium, with a portable helium mass spectrometer as the detector. In addition to the braze joints, various other areas of the propulsion subsystem such as the service valves and the rocket engine assembly flex hoses were leak-checked at working pressure.

B. Functional Tests

The purpose of the propulsion subsystem functional test can be divided into four main categories:

(1) To operate each component of the subsystem without destroying the integrity of one-time use items such as squib valves, burst disks, etc.

(2) To observe any possible interaction between components when they are operating under normal conditions.

(3) To verify that all subassembly components meet their flight performance criteria.

(4) To provide assurance that the functional operation of a component has not been compromised as a result of other subsystem tests such as vibration.

The various portions of the functional test are:

(1) Regulator lockup test

(2) Relief valve assembly functional test

(3) Service valves leak test

(4) Check valves cracking pressure and leak test

(5) Rocket engine assembly valve dual coil resistance test

(6) Rocket engine assembly valve actuation test

(7) Injector orifice flow survey

(8) Rocket engine assembly valve seat leak test

(9) Gimbal actuator functional test

C. Vacuum Chamber Leak Test

The purpose of the vacuum chamber tests was to verify that the propulsion subsystem total external leakage was no greater than 1×10^{-3} STP cm^3/s when the chamber was pressurized with helium at working pressure. A secondary purpose of the test was to verify that outgassing of various components on the subsystem, such as cabling, was within acceptable limits. Although the two propellant tanks and the pressurant tanks were pressurized, no attempt was made to pressurize the feed lines to the REA since these lines normally contain only liquid propellant. Furthermore, helium gas would soon have permeated the Teflon lining of the flex hoses and obscured the test results.

Table 25 contains a summary of subsystem acceptance test problems and solutions.

D. Flight Preparations

The rebuilt engineering test model was shipped to AFETR and used for a complete exercise of the prelaunch operations to be performed on the flight subsystems. The rebuilding of the ETM included replacing propellant bladders with new bladders. The check and relief valves were replaced, and the expended pyrotechnic valve manifolds were modified to allow testing and propellant loading without placing propellants at the engine valve inlets. Leak testing of the subsystem was performed to assure that all rework operations were successfully accomplished.

In support of the pathfinder operation at AFETR, the necessary support and facility equipment was assembled and installed. The propellant service trailers were validated, the pneumatic control console was installed in a trailer and validated, and all gas servicing lines were installed to the propellant loading building.

Operation and testing procedures were modified to reflect the facilities and equipment differences and submitted to Range and Pad Safety for review and approval.

A typical sequence of testing was conducted which included helium leak test, functional test, squib installation and propellant loading operations, propellant unloading, and vacuum drying of the subsystem. All the procedures and support equipment, as well as facilities to be used in flight operations, were successfully employed. As a result of performing the operations on the pathfinder subsystem and conducting the propellant loading operations, many modifications were made in the formal procedures for use during operations with the actual flight subsystems.

The pathfinder subsystem was later used with the PTM spacecraft for launch vehicle interface testing. All the testing was successful and provided an excellent proving ground before the conduct of the prelaunch preparations on the flight systems.

The preparations of the propulsion subsystems for launch at AFETR can be divided into three main areas:

(1) Performance of a subsystem leak test similar to the proof and leak test conducted at JPL but without taking the subsystem to the proof pressure levels.

(2) Repeat of the propulsion subsystem functional test.

(3) Installation of pyrotechnics, fuel and oxidizer fill, and pressurization of the subsystem.

Prior to installation of pyrotechnics onto the subsystem, an initial checkout of the squibs was performed. This checkout measured the resistance of each bridgewire and verified proper dielectric resistance from bridgewire to bridgewire and from bridgewire to squib case.

After verifying that each propulsion subsystem pyrotechnic valve ram was properly seated, a squib with new O-ring was installed and torqued into each valve. Upon installation of all the squibs, the pyrotechnic field test kit was attached to the main pyrotechnic harness connector. As each connector was attached to its assigned squib, the field test kit indicated that proper contact had been made and that cable plus squib resistance valves were suitable for flight.

The weighing scale platform provided a convenient access for leak and functional tests. At the conclusion of these tests, the subsystem was in a position to proceed directly into propellant loading.

The propellant loading procedure can be divided into seven main operations:

(1) Evacuation of the propellant tank

(2) Propellant loading

(3) Propellant backflow

(4) Gas entrainment measuring

(5) Propellant precision downloading

(6) Pressurization

(7) Propellant tank X-ray

In addition to the two flight subsystems, the PTM subsystem was also fueled and pressurized and maintained in readiness as a spare. All loading operations were conducted in full SCAPE suits (Fig. 27).

After preparation, the subsystem pressures were monitored with the transducer monitoring console and were recorded by a sampling printer. In addition, a toxic vapor detector was used to detect any possible propellant leakage.

After installation onto the spacecraft, pressure monitoring was accomplished through the spacecraft telemetry system at regular intervals prior to launch. Toxic vapor detector monitoring was also continued up through launch.

Two significant problems arose during flight preparation, both involving the oxidizer propellant loading. At temperatures of 24°C (75°F) and above, the vapor pressure of the oxidizer was sufficiently high so that when the service trailer valve (located next to the sight glass used to confirm bubble-free flow) was opened, bubbles were formed creating the impression that the subsystem propellant tank was insufficiently bled of gas. At lower oxidizer temperatures of 19 to 20°C (67 to 68°F), the only bubbles observed were those being forced from the subsystem tank during the overflow process. During the use of the gas entrainment measuring device (GED), the oxidizer in the GED would warm up and, by expanding, tend to change the column height in the device. Readings taken to measure the quantity of gas entrained in the propellant tank had to be taken relatively quickly to prevent erroneous results.

VIII. PROPULSION SUBSYSTEM SUPPORT EQUIPMENT

Figure 28 shows the major items of propulsion support equipment. The support equipment required for various operations is listed in Table 26. The various major pieces of support equipment are described briefly below.

A. Pneumatic Control Console

The pneumatic control console (PCC) was manufactured by Airesearch Manufacturing Division, The Garrett Corporation, for Hughes Aircraft Corporation during the Surveyor program and was used in the same configuration during the Mariner Venus 1967 and Mariner Mars 1969 programs. To adapt the PCC to MM '71 functions, the internal circuitry was rearranged. The modifications and redesign allowed the PCC to be used at JPL, Edwards Test Station, and AFETR as the primary pressure control during subsystem PSS leak check, proof test, and functional test and to pressurize the propellant and solvent loading trailers.

B. Auxiliary Pneumatic Control Console

The auxiliary pneumatic control console was used to perform a leak test of the propellant tank bladder where accuracy in the range from zero to 1.38×10^5 N/m^2 (20 psi) is required. The console provided for a single input and two regulator outputs.

C. Fuel Service Trailer

The function of the fuel trailer was to accurately and safely load the propulsion system with a precise amount of filtered, vapor-free fuel (MMH). The trailer, also surplus from the Surveyor program, includes a vacuum system, toxic gas processing system, nitrogen gas processing system, gas purge system, and a propellant transfer system.

In order to use the fuel trailers for MM '71, the service trailer body and chassis assembly was rebuilt to support the additional weight. In the new configuration, the trailer was extended and fitted with a four-wheel suspension. A new circulation system and a 0.265-m^3 (70-gal) supply tank were installed. Trailer functions were expanded to include an entrained gas measuring device, flex hose aspiration circuit, and propellant bulk temperature indicator. A new control panel was also installed, and all test and service sequences were performed by operating valves mounted on this panel. All applicable components that could be salvaged from the Surveyor propellant service trailers were reused after cleaning.

D. Oxidizer Service Trailer

The design of this trailer was essentially the same as for the fuel service trailer, except that it was configured for use with the oxidizer, nitrogen tetroxide. The oxidizer service trailer was rebuilt for MM '71, similar to the rebuilding of the fuel service trailer.

E. Fuel Flush Cart

The fuel vacuum purge and flush cart functioned to reduce the residual fuel (MMH) to a safe level in the propulsion system. The flushing cart was used at any time it was necessary to remove fuel from the propulsion subsystem. A secondary use was to clean the fuel servicing trailer. Complete propellant removal from the propulsion system or the fuel service unit required repetitive use of isopropyl alcohol to flush the system and alternate helium purge and evacuation cycles.

In order to adapt the Surveyor surplus flush carts for the Mariner 1971 propulsion subsystem, all valves, regulators, gages, vacuum pumps, tank, etc., were removed, cleaned, and, where possible, reused. A larger circulation system was required and installed, with a solvent load volume of 0.265 m^3 (70 gal). A new control panel was also installed and all test and service sequences were performed by operating valves mounted on this panel.

F. Oxidizer Flush Cart

The design of the oxidizer flush cart was essentially the same as that for the fuel flush cart except that the oxidizer cart was configured for use with the solvent Freon. The oxidizer flush cart was rebuilt for MM '71, similar to the rebuilding of the fuel flush cart.

G. Solvent Service Cart

The solvent service cart was designed to load and unload the subsystem with referee fluids and to vacuum-dry the PSS. The cart consisted of two separate solvent service systems – one for the oxidizer system and one for the fuel system. The oxidizer system used Freon solvent and the fuel system used isopropyl alcohol solvent. The liquid capacity was 0.114 m^3 (30 gal) with 10-μm filtration to maintain cleanliness. Test and service sequences were accomplished by operating manual valves.

H. Handling Fixture

The subsystem handling fixture was an aluminum ring approximately 57 inches in diameter with a 10.16 by 15.24 cm (4 by 6 in.) cross section. Its flight hardware interface details were identical to spacecraft features. For attachment of the handling fixture to all other handling equipment, the fixture provided three ball jointed brackets equally spaced on the perimeter of the ring. The attachment feature was standard on all subsystem handling equipment.

I. Workstand

The workstand, originally a surplus Ranger dolly, was fitted with the three-point provisions which were standard on all PSS handling equipment. Mounted on new high-quality hard rubber casters, the complete workstand was approximately 28.58 cm (12 in.) high. Installed on the workstand, the PSS was at a convenient level to accomplish many of the assembly test and operation sequences.

J. Transducer Monitoring Console

The transducer monitoring console consisted of a single bay unit containing the circuitry required to monitor the subsystem pressure and temperature parameters during checkout and test, and after propellant loading. A front panel contained four digital voltmeters and rotary parameter selector switches. The digital readouts were in engineering units.

Up to three propulsion subsystems could be connected simultaneously to the console. A small remote box was assigned to each subsystem. The remote box was permanently wired into the transducer monitoring cable and located near its subsystem. Standardizing resistors located in the remote box were used to normalize the temperature transducer outputs. The connectors on the transducer monitoring console cables were flight-qualified.

K. Engine Valve Actuation Console

The engine valve actuation console consisted of a single bay unit containing the power supplies and control circuitry to actuate the PSS engine valve. The regulated power supply delivered 30 ±0.01 V to either engine valve coil No. 1 or coil No. 2 or both together when operated manually or automatically. Automatic actuation of the engine valve coils provides pulsed operation from 50 to 200 ms so that both opening and closing transients can be displayed on a single oscilloscope picture.

L. Transporter

The subsystem transporter was designed and manufactured for the Surveyor program. In order to use the Surveyor transporter for MM '71, a new adapter to accommodate the standard three-point mounting system was installed. The inner shroud was eliminated and the steering gear was modified to eliminate damaging the steering tongue when cornering.

M. Rotation Fixture

The rotation fixture was designed for use exclusively during tests at Edwards Test Station. At Edwards, the subsystem was delivered to the test stand fully loaded with the engine nozzle up; however, the firings were to be done with the nozzle down. The rotation fixture performed the required reorientation. Originally a surplus aircraft engine repair stand, the rotation fixture was reconditioned and fitted with a large cradle weldment. The rotation was accomplished through the heavy-duty worm gear train of the aircraft engine repair stand by means of an added electric motor and gear reducer. The operation was controlled remotely and progressed at about 1/4 revolution/minute.

N. Low-Level Positioner

The low-level positioner was used to provide good access to the subsystem for brazing and X-ray. Designed and built for assembly and test of entire spacecraft of the earlier Mariner series, the low-level positioner was ideally suited to position the subsystem for access of the brazing tools and X-ray head.

O. Data Logger

The data logger system was used to monitor subsystem pressures after propellant loading at AFETR. The system uses the output of the transducer monitoring console and periodically samples and prints data. This procedure provided about one sample of each pressure each hour after propellant loading.

P. Toxic Vapor Detector

The toxic vapor detectors used were of the Teledyne wet cell type. These units are portable and were connected to provide both local and remote alarms. The units were adequate but prone to frequent false alarms and required frequent maintenance when used continuously.

Q. Propulsion Simulator

The propulsion simulator was mounted on the spacecraft when the propulsion subsystem was not available. The simulator shown on Fig. 29 contains potentiometers to simulate each telemetry sponsor, a propellant valve, and mounting provisions for a pair of gimbal actuators.

R. Engine Injector Scanner

The engine injector scanner is used to determine that each injector orifice is not obstructed and will allow full propellant flow. To make the scan, the engine valve is opened and nitrogen gas is flowed through the orifices. The scanner employs a small heated wire which is moved over the injector orifice openings. The resistance of the heated wire is measured as the wire is moved around the ring of injector orifices. Cooling of the wire occurs when it is exposed to the gas flowing from an orifice and the resistance change is recorded.

NOMENCLATURE

AFETR	Air Force Eastern Test Range
AISI	American Iron and Steel Institute
AN	Air Force/Navy Aeronautical
BLC	boundary-layer cooled
C_f	thrust coefficient
CRES	corrosion resistant steel
ETM	engineering test model
DSM	development system model
ETS	Edwards Test Station
FA	flight approval
FEP	fluorinated ethylene propylene
PIA	propellant isolation assembly
PSS	propulsion subsystem
PTM	proof test model
REA	rocket engine assembly
SCAPE	self-contained atmospheric protection ensemble
STP	standard temperature and pressure
S/S	slam start
TA	type approval
TFE	tetrafluoroethylene
TIG	tungsten inert gas
GED	gas entrainment measuring device
MDC	mission duty cycle
MLT	margin limit test
MMC	Martin Marietta Corporation, Denver Division
MMH	monomethylhydrazine

MR	mixture ratio
O/F	oxidizer to fuel ratio
P_c	chamber pressure
PCA	pressurant control assembly
PCRA	pressure check and relief assembly

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Table 1. Required Propulsion Subsystem Telemetry Outputs

Parameter	Range
Nitrogen tank pressure	0 to 34.5×10^6 N/m^2 (0 to 5000 psia)
Propellant tank and line pressures	0 to 2.8×10^6 N/m^2 (0 to 400 psia)
Thrust chamber pressure	0 to 1.4×10^6 N/m^2 (0 to 200 psia)
Nitrogen tank temperature	-45 to 54°C (-50 to 130°F)
Propellant tank temperatures	-7 to 64°C (+20 to 147°F)
Engine valve temperature	-7 to 149°C (+20 to 300°F)
Engine injector temperature	-7 to 82°C (+20 to 180°F)

Table 2. Propulsion Subsystem Performance Characteristics

Parameter	Value	
Vacuum thrust	1334 ±89 N	(300 ±20 lb_f)
Vacuum specific impulse	2775 ±49 m/s	(283 ±5 lb_f s/lb_m)
Thrust chamber expansion ratio		40:1
Thrust chamber pressure	$806.7 \times 10^3 \pm 55 \times 10^3$ N/m^2	(117 ±8 psia)
Propellant loaded mixture ratio, O/F by weight[a]		$1.50^{+0.05}_{-0.03}$
Nominal oxidizer flow rate	0.289 kg/s	(0.637 lb_m/s)
Nominal fuel flow rate	0.192 kg/s	(0.424 lb_m/s)
Propellant load capacity	462.7 kg	(1020 lb_m)
Usable propellant load capacity	440 kg	(970 lb_m)
Propellant loading accuracy	±0.45 kg	(±1.0 lb_m)
Minimum burn duration		0.4 s
Shutdown impulse variation, 3σ	±22.2 N/s	(±5 lb_f-s)

[a]O = oxidizer; F = fuel.

Table 3. Mariner Mars 1971 Propulsion Weight Summary (Dry Mass)

Component	Dry Mass		Assembly Total Dry Mass	
	kg	lb_m	kg	lb_m
Propulsion Subsystem				
Pressurant tank assembly			24.9	55.0
Pressurant tanks (2)	24.72	54.50		
Temperature transducer	0.04	0.10		
Transition tube	0.18	0.40		
Pressurant control assembly			5.6	12.4
Pressure transducer	0.18	0.40		
Pyrotechnic valve assembly	4.17	9.20		
Pressurant filter	0.45	1.00		
Pressure regulator	0.82	1.80		
Pressure check-relief assembly (2)			2.1	4.6
Check valve	0.18	0.40		
Relief valve	0.61	1.35		
Service valve assembly	0.11	0.25		
Pressure transducer	0.14	0.30		
Propellant tank assembly (2)			30.4	67.0
Propellant tank	10.18	22.45		
Bladder	1.22	2.70		
Standpipe assembly	3.63	8.00		
Service valve assembly	0.11	0.25		
Temperature transducer	0.04	0.10		
Propellant isolation assembly (2)			10.2	22.4
Pyrotechnic valve assembly	4.17	9.20		
Propellant filter	0.77	1.70		
Pressure transducer	0.14	0.30		
Rocket engine assembly			7.8	17.1
Tubing and fittings			5.9	13.1
Total			86.9	191.6
Propulsion assembly				
Thrust Plate Assy			5.72	12.6
Truss and Ring Assy			9.12	20.1
Propellant Tank Thermal Covers (2)			2.5	5.5
Cable Harnesses (2)			2.2	4.9
Gimbal Actuators (2)			2.5	5.6
Squibs (15)			1.0	2.3
Total			23.04	51.0
Total propulsion subsystem and assembly			110.0	242.6

Table 4. Nominal Pressure Budget and Performance Summary, TA Propulsion Subsystem

Item	Oxidizer	Fuel
Regulator outlet pressure, N/m^2 (psi)	1740.94×10^3 (252.50)	
Check valve pressure drop, N/m^2 (psi)	17.24×10^3 (2.50)	17.24×10^3 (2.50)
Bladder pressure drop, N/m^2 (psi)	0.00 (0.00)	0.00 (0.00)
Tank liquid pressure, N/m^2 (psi)	172.37 (250.00)	172.37 (250.00)
Standpipe and outlet pressure drop, N/m^2 (psi)	19.72 (2.68)	12.13 (1.76)
Squib and line pressure drop, N/m^2 (psi)	58.95 (8.55)	53.02 (7.69)
Filter pressure drop, N/m^2 (psi)	26.34 (3.82)	17.24 (2.50)
Engine inlet pressure, N/m^2 (psi)	1620.00 (234.96)	1641.31 (238.05)
Valve pressure drop, N/m^2 (psi)	194.85 (28.26)	194.85 (28.26)
Injector pressure drop, N/m^2 (psi)	367.70 (53.33)	344.12 (49.91)
Calibration orifice pressure, N/m^2 (psi)	266.14×10^3 (38.60)	308.82×10^3 (44.79)
Chamber pressure, N/m^2 (psia)	791.11×10^3 (114.74)	
Propellant flowrate, kg/s (lb_m/s)	0.2852 (0.6287)	0.1827 (0.4027)
Total propellant flowrate, kg/s (lb_m/s)	0.4678 (1.0314)	
Mixture ratio, O/F	1.561	
Thrust, N (lb_f)	1316.6 (296.00)	
Specific impulse, m/sec (lb_f-s/lb_m)	2814.31 (286.98)	
Throat area, cm^2 ($in.^2$)	9.29 (1.440)	
Nozzle area ratio	40.00	

Table 5. Mariner 9 Performance Prediction Summary, Spacecraft Mass and Propulsion Capability

Item	Spacecraft Mass	
	kg	lb_m
Initial spacecraft mass M_0		
Spacecraft dry separated mass	505.0	1113.4
Attitude control nitrogen	2.4	5.4
Propulsion nitrogen	14.6	32.2
Oxidizer (N_2O_4)	290.3	640.0
Fuel (MMH)	186.0	410.0
Total M_0	998.4	2201.0
Final spacecraft mass M_f		
Spacecraft dry separated mass	505.0	(1113.4)
Attitude control nitrogen	1.8	(4.0)
Propulsion nitrogen	14.6	(32.2)
Oxidizer	15.6	(34.5)
Performance reserve	3.8	(8.5)
Holdup	11.8	(26.0)
Fuel	12.2	(26.8)
Performance reserve	5.8	(12.8)
Holdup	6.8	(14.9)
Total M_f	549.2	(1210.9)
Propulsion capability		
Specific impulse:		
I_s = 2821.57 m/s		
(287.72 lb_f-s/lb_m)		
$\Delta V = I_s \ln \frac{M_0}{M_f} = 1683$ m/s		

Table 6. Development System Model Test History

DSM Test No.	Run No.	Date	Event	Comments
1	Dd 383	5-2-69	30-s engine firing	Breadboard system
2	Dd 384	5-16-69	Regulator slam start (S/S)	
	Dd 385	5-21-69	100-s engine firing	Gimbal investigation
	Dd 386	5-22-69	Line surges	
	Dd 388	5-29-69	Regulator S/S	Simulated tank volume
	Dd 390		Line surges	
3	Dd 392	6-27-69	100-s engine firing	Saturated oxidizer
4	Dv 312	1-20-70	Engine gas blowdown	Gas resistance check
	Dv 313	1-23-70	O1, F1 opening, regulator S/S	
	Dv 314	1-23-70	16-s engine firing	
	Dv 315	1-27-70	Regulator S/S	
	Dv 316	1-28-70	Regulator S/S, Line surges, 62-s engine firing	To 73% expulsion
	Dv 317	1-29-70	O2, F2 closing	
5	Dv 318	3-5-70	Engine gas blowdown	Gas resistance check
	Dv 319	3-6-70	Regulator S/S, O3, F3 opening	724×10^3 N/m^2 (105 psi), fuel transducer shift
	Dv 320	3-6-70	100-s engine firing	-1°C (30°F) propellants
	Dv 321	3-10-70	Regulator S/S, 700 s engine firing	To 83% expulsion
	Dv 322	3-11-70	O4, F4, closing, "hammer test"	Lines overpressurized
6	Dv 323	3-19-70	Regulator S/S, O5, F5 opening	Gas leakage
	Dv 324	3-19-70	0.4-s, 871-s engine firings	Gimballed to 10 deg
	Dv 325	3-23-70	Regulator S/S	Lines plugged at tanks
	Dv 326	3-31-70	Regulator S/S	Lines plugged at tanks

Table 7. Engineering Test Model Firing Summary

Series	Run No.	Date	Events	Fuel Temperature °F	Fuel Temperature °C	Oxidizer Temperature °F	Oxidizer Temperature °C	Burn Duration, s	Saturation
1		5/20	Propellant loading	61	16.1	61	16.1		None
	Dv 327	5/21	O1-F1-P1 actuation	69	20.6	69	20.6		
	Dv 328	5/21	Firing	67	19.4	72	22.2	4.04	
	Dv 329	5/22	Firing	70	21.1	69	20.6	909.33	
	Dv 330	5/22	Post-firing soak	85	29.4	82	27.8		
	Dv 331	5/22	Post-firing soak	85	29.4	82	27.8		
2		6/16	Propellant loading	32	0	32	0		None
	Dv 332	6/16	P2 actuation	43	6.1	46	7.8		
	Dv 333	6/16	P3 actuation	43	6.1	47	8.3		
	Dv 334	6/16	Firing	40	4.4	44	6.7	901.04	
	Dv 335	6/16	Post-firing soak	77	25.0	77	25.0		
	Dv 336	6/17	O2-F2 actuation	82	27.8	82	27.8		
	Dv 337	6/18	O3-F3 actuation	74	23.3	74	23.3		
	Dv 338	6/18	Firings	66	18.9	68	20.0	0.396 and 15.71	
3		6/25	Propellant loading	31	-0.5	32	0		1724×10^3 N/m^2 (250 psi)
	Dv 339	6/25	Firings	38	3.3	41	5.0	8.27, 0.41, and 873.97	
	Dv 340	6/26	Firings	82	27.8	78	25.6	8.02 and 5.83	1724×10^3 N/m^2 (250 psi)
4		7/1	Propellant loading	98	36.7	94	34.4		
	Dv 341	7/1	Firing	96	35.6	90	32.2	900.76	None
	Dv 342	7/6	O4-F4 actuation	98	36.7	98	36.7		
	Dv 343	7/7	O5-F5 actuation	85	29.4	85	29.4		
	Dv 344	7/7	Firing	83	28.3	83	28.3	9.97	675×10^3 N/m^2 (98 psi)
	Dv 345	7/7	P-4 actuation	93	33.9	93	33.9		
5			Propellant loading						None
	F 63	8/27	Firing	93	33.9	93	33.9	11.798	
	F 64	11/18	Firing	56	13.3	56	13.3	30.918	
	F 65	11/18	Firing	58	14.4	58	14.4	870.422	
	F 66	11/20	P5 actuation, firing						

Total number of burns = 16
Total accumulated burn time, s = 4497.23

Table 8. Engineering Test Model Subsystem Engine Test Performance

Run No.	Test Duration, s	Average Chamber Pressure, N/m^2 (psia)	Average Mixture Ratio (O/F)	Average Vacuum Thrust,[a] N (lb_f)	Average Vacuum Specific Impulse, m/s ($lb_f\ s/lb_m$)	Average Characteristic Velocity, m/s (ft/s)	Average Propellant Temperature, °C (°F)	Comments
Dv 329	909.33	813.6×10^3 (118.0)	1.60	1354.0 (304.4)	2825.3 (288.1)	1578 (5177)	21 (70)	
Dv 334	901.04	813.6×10^3 (118.0)	1.61	1354.0 (304.4)	2804.7 (286.0)	1566 (5138)	5 (42)	
Dv 339	873.97	795.6×10^3 (115.4)	1.55	1324.2 (297.7)	2777.2 (283.2)	1551 (5090)	5 (40)	Saturated propellants
Dv 341	900.76	803.2×10^3 (116.5)	1.58	1336.7 (300.5)	2825.3 (288.1)	1578 (5178)	34 (93)	Pressurant depletion
F 65	870.42	852.2×10^3 (123.6)[b]	1.59[b]	1418.1 (318.8)[b]	2771.4 (282.6)[b]	1548 (5080)[b]	14 (58)	Atmospheric pressure

[a]Calculated from chamber pressure using $C_f = 1.790$.

[b]Sea level engine installed.
Manufacturer's P_c calibration unknown; assumed to be linear.
High P_c is due to high inlet pressures; regulator referencing atmospheric pressure.

Table 9. Type Approval Test Series 1

Test event	Simulated Flight Event
Propellant vibration (3 axes)	Launch
Installation in vacuum chamber	2-week coast
Moog valve open	Moog valve open
P1, O1, F1 open	P1, O1, F1 open
8-s burn	Midcourse burn
1-day coast	1-week coast
P2 close	P2 close
O2, F2 close	O2, F2 close
Check valve test	
1-day coast	6-month coast
O3, F3 open	O3, F3 open
P3 open	P3 open
10-s burn	Midcourse burn
1-day coast	3-week coast
900-s burn	Orbit insertion burn
2-day coast	2 to 4-day coast
0.4-s burn	Orbit trim burn
3-day coast	2 to 4-day coast
40-s burn	Orbit trim burn
---	Coast
---	Close P4, O4, F4
---	Orbit planet

Table 10. Type Approval Test Series 2

Test Event	Simulated Flight Event
900-s burn	Orbit insertion burn
3-day coast	2 to 4-day coast
16-s burn	Orbit trim burn
1-day coast	2 to 4-day coast
---	Orbit trim burn
---	Coast
P4, F4, O4 close	Close P4, O4, F4
1-day coast	Orbit planet
O5, F5 open	---
P5 open	---
10-s burn	---
1-day coast	---

Table 11. Summary of Qualification Test Requirements for Explosive Valves

Test	Description
Temperature	-54°C to +52°C (-65°F to +125°F) with 1-h soak at 66°C (150°F) prior to firing at 52°C (125°F)
Vibration	18.1 g rms (random, overall)
Shock	200 g saw tooth
Firing	Two normally open units at 18°C (65°F) using GN_2 at 27.6×10^6 N/m^2 (4000 psig); two normally closed units at -54°C (-65°F) using GN_2 at 27.6×10^6 N/m^2 (4000 psig); two normally closed and two normally open units at 52°C (125°F) using water at 1620×10^3 N/m^2 (235 psig); six normally closed and two normally open units at 52°C (125°F) using GN_2 at 27.6×10^6 N/m^2 (4000 psig); two normally closed and two normally open units at -7°C (20°F) using water at 1620×10^3 N/m^2 (235 psig); twelve normally closed and twelve normally open units at ambient conditions.
Burst pressure	55.2×10^6 N/m^2 (8000 psig) (hydrostatically) Note: As an optional test, four normally closed and four normally open units were subjected to burst pressure prior to firing.
Leakage	This test was performed as a success criterion after exposure to each environment in accordance with the above FA and lot acceptance requirements.

Table 12. Pressurant Relief Valve Margin Limit Test Program

Test Condition	Valve Serial Number			
	8	9	10	11
Acceptance test at vendor	1[a]	1	1	1
Abbreviated acceptance test at JPL consisting of: proof at 3930×10^3 N/m^2 (570 psig), leak, crack, and reseat	2	2	2	2
Diaphragm endurance, 2000 cycles, 0-2034 $\times 10^3$ N/m^2 (0-295 psig)	3	4		
Diaphragm reverse loading, 4 cycles, 1 h each, 0-2413 $\times 10^3$ N/m^2 (0-350 psig)	4	3		0
Diaphragm rupture	5	5	6	6
Vibration: sine and random combined; 3 axes; TA level	6	6	4	4
Shock loading, 200 g, 3 axes			5	5
Valve poppet endurance, 5000 cycles	7	7		
Flow vs ΔP, 50 to 150%, room temperature	8	8	7	7
Flow vs ΔP, high, 43.3°C (110°F) and low, -1.1°C (30°F) temperature; at rated flow	9	9		
30-day propellant exposure, N_2O_4			3	
30-day propellant exposure, MMH				3

[a]Numbers in the columns under each valve serial number indicate the sequential order of testing.

Table 13. Mariner Mars 1971 Propellant Tank Design Requirements

Item	Requirement
Material	6A14V titanium
Shape	Spherical
Diameter	74.9 cm (29.5 in.)
Volume	219,600 cm^3 (13,400 $in.^3$, minimum)
Weight (tank only)	9.98 kg (22 lb_m)
Wall thickness	0.084 $^{+013}_{-000}$ cm (0.033 $^{+005}_{-000}$ in.)
Operating pressure (nominal)	2068×10^3 N/m^2 (300 psig)
Room temperature proof pressure	4136×10^3 N/m^2 (600 psig)
Liquid N_2 proof pressure	5792×10^3 N/m^2 (840 psig)
Room temperature burst (actual)	5557×10^3 N/m^2 (806 psig)
Helium leak at 2068×10^3 N/m^2	Less than 1×10^{-7} STP cm^3/s
Heat treatment	1.14×10^9 N/m^2 to 1.21×10^9 N/m^2 (160,000 to 175,000 psi)
Tank opening	17.8 cm (7-in.-diam.) bolted flange

Table 14. Summary of Results of Bladder Material Testing

Item		TFE/FEP T-120 laminate	TFE/FEP TE-9511 codispersion
Uniaxial strain (wet) after 152 h in heptane		Ultimate tensile reduced 30%	Ultimate tensile unchanged
		Ultimate elongation reduced 50%	Ultimate elongation increased 10%
Uniaxial fatigue (dry) yield stress cycles to failure		7,000	20,000
Biaxial strain at rupture (dry), %		40	125
Biaxial strain at rupture (wet) after 150 h in solution, %	Isopropyl alcohol	6	95
	Freon	5	128
	Heptane	2.5	70

Table 15. Mariner Mars 1971 Propellant Acquisition Standpipe Primary Design Requirements

Item	Requirement
Propellant flow rates	
N_2O_4	0.27 kg/s (0.64 lb_m/s)
MMH	0.18 kg/s (0.40 lb_m/s)
Propellant temperatures	
Prelaunch and launch	15.6-104.4°C (60-90°F)
In-flight--zero g	4.4°C-104.4°C (40-90°F)
Acceleration loads	
Prelaunch and launch	1 to 8 g (negative)
In-flight space operation	0.14 to 0.3 g (positive)
Minimum expulsion duration	
1st start	2.0 s
Subsequent starts	0.4 s
Pressure drop at nominal flow	20.7×10^3 N/m^2 (3 psid) (max)
Ground handling attitudes tipping angle:	±45 deg
Expulsion efficiency:	98%

Table 16. Component and Assembly Tests, MM '71 Standpipe Assembly/Propulsion Subsystem

Test Type
Cone structural evaluation--column collapse mode
Bladder material strength limits--as loaded against perforated cone
Expulsion efficiency
Pressure drop vs flow rate
Trap volume calibration
Gas retention capability--using isopropyl alcohol, Freon, and N_2O_4
Surge flow (water hammer) effects on gas critical height
Vacuum fill procedures development

Table 17. Sequence of Hose Assembly Type Approval Tests

Test Sequence	Test Performed on Specimens as Shown Below	
	S/N 108	S/N 110
Pressure drop	X	X
Shock	X	X
Examination - physical	X	X
Examination - radiographic	X	X
Leak test	X	X
Pressure drop	X	X
Flexure force	X	X
Static acceleration	X	X
Examination - physical	X	X
Examination - radiographic	X	X
Leak Test	X	X
Vibration	X	X
Examination - physical	X	X
Examination - radiographic	X	X
Leak test	X	X
Pressure drop	X	X
Flexure force	X	X
Flexure/life	X	X
Flexure force	X	X
Examination - physical	X	X
Examination - radiographic	X	X
Leak test	X	X
Pressure drop	X	X
Minimum burst	X	X
Leak test	X	X
Examination - physical	X	X
Examination - radiographic	X	X
Ultimate burst		X
Disassembly and inspection	X	X

Table 18. Engine Operating Conditions

Parameter	Minimum	Standard	Maximum
Environmental temperature, °C (°F)	1.1 (30)	21 (70)	32 (90)
Environmental altitude pressure, N/m^2 (psia)	0	0	1.1×10^3 (0.16)[a]
Propellant inlet temperature, °C (°F)	0 (30)	21 (70)	32 (90)
Oxidizer or fuel differential, °C (°F)	0	0	5 (10)
Oxidizer inlet pressure, N/m^2 (psia)	1586×10^3 (230)	1634×10^3 (237)	1689×10^3 (245)[b]
Fuel inlet pressure, N/m^2 (psia)	1613×10^3 (234)[b]	1668×10^3 (242)	1724×10^3 (250)[b]
Inlet pressure differential (fuel-oxidizer), N/m^2 (psi)	0	34.5×10^3 (5)	62×10^3 (9)
Propellant valve applied voltage, Vdc	28	30	32
Propellant saturation level			
Oxidizer, STP cm^3/g at 21°C	0	0	2.50
Fuel, at 21°C	0	0	0.60

[a]Environmental altitude pressure is applicable only as a test condition. REA performance and associated tolerances are based on operation in a vacuum.

[b]Minimum/maximum inlets due to regulator outlet pressure band such that minimum/maximum inlet pressures cannot occur simultaneously.

Table 19. Engine Performance Requirements

Item	Requirement
Thrust, vacuum, N (lb_f)	1334 ±89 (300 ±20)
Mixture ratio, O/F (by weight)	$1.57\ {}^{+0.05}_{-0.20}$
Specific impulse, vacuum minimum, m/s $\left(\frac{lb_f\text{-s}}{lb_m}\right)$	2775 (283)
Shutdown impulse, vacuum, N-s (lb_f-s) (95% coverage, 90% confidence)	31 ±22 (7 ±5)
Roll torque, max, cm-N (in. -lb_f)	16.9 (1.5)
Lifetime	2 mission duty cycles[a]

[a]A mission duty cycle is defined as five starts for an accumulated total firing time of 1000 s spread over 200 days, with the longest steady state firing duration being 940 s, the shortest commanded firing duration being 0.40 s, with a minimum of 24 h between consecutive engine starts.

Table 20. Rocket Engine Assembly Prequalification Test Summary

Test Sequence	Engine S/N 0003	Completion Date	Engine S/N 0004	Completion Date
Flight acceptance (not a part of prequalification)	X	12-15-69	X	12-15-69
Cold-flow verification	X	12-29-69	X	12-29-69
Vibration (sinusoidal and random -- TA level)	X	1-7-70	X	1-7-70
Visual inspection and leak and functional tests	X	1-15-70		
Cold-flow verification	X	1-16-70		
Propellant saturation and analysis procedure study	No engine involved in study -- 1-19-70			
Hot fire to cold flow calibration verification	X	11-23-70		
Mission duty cycle test and minimum impulse tests	X	1-26-70		
Visual inspection and external leak test	X	1-27-70		
Performance survey tests	X	2-5-70		
Leak and functional tests	X	2-13-70		

Table 21. Engine S/N 0003 Prequalification Firing Test Summary

Test Description	Test Duration (No. of Tests)	Propellant Temperature, °C (°F)		Propellant Saturation, cm^3/g	
		Oxid.	Fuel	Oxid.	Fuel
Hot fire to cold flow calibration verification	10-s duration each (4 tests)	13.3 (56)	13.3 (56)	0	0
Mission duty cycle					
(1) 2nd midcourse correction	4 s (1 test)	6.6 (44)	6.1 (43)	2.4	0.5
(2) Pulse tests	15 pulses 0.4-s duration			2.4	0.5
(3) 1st orbit trim	11 s (1 test)	-2.2 (28)	-1.1 (30)	2.4	0.5
(4) 1st midcourse correction	12 s (1 test)	-2.8 (27)	-1.1 (30)	2.4	0.5
(5) 2nd orbit trim	13 s (1 test)	-3.3 (26)	-1.1 (30)	2.4	0.5
(6) Orbit insertion	861 s (1 test)	-3.9 (25)	-1.1 (30)	2.2	0.3
Performance survey	800 s (1 test)	-1.7 (29)	0 (32)	0	0
Performance survey	800 s (1 test)	-3.9 (25)	-1.1 (30)	2.1	0.4
Orbit insertion	861 s (1 test)	28.3 (82)	31.7 (89)	2.7	0.7
Performance survey	787 s (1 test)	29.4 (85)	32.8 (91)	2.7	0.7
Performance survey	800 s (1 test)	25.6 (78)	27.2 (81)	0	0
Performance survey	800 s (1 test)	28.3 (83)	31.1 (88)	0	0

Table 22. Rocket Engine Assembly Type Approval Test Sequence

TA Unit S/N 0004	TA Unit S/N 0005
Refurbishment with FA leak and functional	Flight acceptance
Cold flow calibration	Humidity
FA and TA vibration	Leak and functional
Final FA leak and functional and calibration verification	TA vibration
Humidity	Leak and functional
Leak and functional	Calibration verification
Midcourse correction firings[a]	Midcourse correction firings
Calibration verification	Orbit insertion burn[c]
MDC sequence 1[b]	External leakage test
Leak and functional	Repeat orbit insertion and complete MDC 1
Calibration verification	Leak and functional
MDC sequence 2	MDC sequence 2
Leak and functional	Leak and functional
Life test	Life test
Leak and functional	Leak and functional
Calibration verification	Calibration verification
Disassembly and inspection	Disassembly and inspection

[a]Invalid due to inadvertent propellant saturation.

[b]Mixture ratio survey and pulses deleted because of nozzle buckling discovered after orbit insertion.

[c]Aborted after 250 s, due to P_c discrepancy.

Table 23. Rocket Engine Assembly Type Approval Program Test Matrix

Test	S/N 0004	S/N 0005
Environmental tests		
Humidity	X	X
Leak and functional	X	X
Sinusoidal vibration	X	X
Random vibration	X	X
Leak and functional	X	X
Cold flow verification	X	X
Hot fire tests at simulated altitude	High saturation	Nominal saturation
MDC 1 at nominal conditions	X	X
Leak and functional	X	X
MDC 2		
Low propellant temperature orbit insertion burn	X	X
High inlet pressure	X	
Low inlet pressure		X
Leak and functional	X	X
Life test		
High propellant temperature	X	X
High inlet pressure	X	X
Leak and functional	X	X
Calibration verification	X	X
Disassembly and inspection	X	X

Table 24. Proof and Leak Test Pressures

Area	Pressure, N/m^2 (psi)	
	Proof	Leak test
Pressurant circuit upstream of explosive valves	41.4×10^6 (6,000)	27.6×10^6 (4,000)
Pressurant circuit downstream of explosive valves	41.4×10^6 (6,000)	27.6×10^6 (4,000)
Pressurant circuit downstream of pressure regulator and upstream of check valves	1758×10^3 (255)[a]	1758×10^3 (255)[a]
Propellant tank circuits downstream of check valves	3103×10^3 (450)	2068×10^3 (300)
Propellant tank bladder, inside to outside	---	28×10^3 (4)
Propellant circuit downstream of explosive valves	3103×10^3 (450)	2068×10^3 (300)
Engine combustion chamber	1724×10^3 (250)	1034×10^3 (150)

[a]Limited because of circuit design (see Fig. 1).

Table 25. Subsystem Acceptance Test Problems

Model	Problem	Solution/Remarks
ETM	During functional test, oxidizer and fuel check valves exhibited excessive leakage.	Propellant tank pressurant bypass lines plugged to minimize probability of liquid propellants at the check valves. Subsequent leak tests indicated significant reduction in leakage.
	Oxidizer tank temperature transducer bond joint failed (observed following initial firing sequence).	Grit paper used to roughen tank surface changed from Mylar-backed to paper-backed. Freon used to clean tank shell surface stored in glass bottles to preclude leaking of contamination from polyethylene storage containers.
TA	Oxidizer bladder leak rate exceeded specification tolerance (6.25×10^{-2} vs 5.0×10^{-2} STP cm^3/s allowable).	Specification changed to allow a leak rate of 6.6×10^{-2} STP cm^3/s helium, maximum. New codispersion laminate bladders procured to replace original design.
	Service valves leakage excessive (greater than 1×10^{-6} STP cm^3/s helium).	No corrective action taken on TA service valves since leak rates were within acceptable limits for test usage.
	Service valve leakage excessive. Degradation of service valve seat.	Revised leak test procedure to remove ground fitting prior to performing service valve leak tests. Valve seat relapped. Torque reduced from 282.4 cm-N (25 in.-lb) to 169.5 cm-N (15 in.-lb) during all prelaunch testing.
	Leakage at 15.87 mm (5/8 in.) fitting of flex line at REA valve.	Aluminum cone seals installed at Flexline-Moog valve interface and at flexline-hangdown 1.27 cm, or 1/2 in. fitting interface.

Table 25 (contd)

Model	Problem	Solution/Remarks
TA (contd)	Loss of pressure in test volume upstream of subsystem regulator indicating possible component internal leak.	Revised test procedure to increase leakage monitoring from 1 to 6 h; instrumentation monitored prior to test for sources of leakage; temperature variations accounted for in determining their effect on pressure changes.
	Extraneous spike observed in closing current trace of engine valve during functional test.	The engine valve actuation console circuitry was modified to isolate the current monitoring circuit from the voltage monitoring circuit.
PTM	PIA (oxidizer and fuel) pressure transducers noise observed at and above 1655×10^3 N/m^2 (240 psia) and 1931×10^3 N/m^2 (280 psia) respectively, during calibration test.	Use as is -- noise observed during transient condition. Flight data taken under steady-state conditions.
	Service valve PB leakage 6.5×10^{-5} STP cm^3/s (Spec: 1×10^{-6} STP cm^3/s). Oxidizer check valve leakage 1.1×10^{-2} STP cm^3/s (Spec: 0.8 STP cm^3/h.)	Valve seat lapped and ceramic ball of smoother surface texture installed. Check valve removed and replaced with check valve.
	Output of engine and oxidizer tank pressure transducer noisy at and above 2034×10^3 N/m^2 (295 psia).	Use as is. Flight data taken at steady-state conditions. Noise observed during transient pressure calibration.
	Check valve S/N 0000033 leakage 673 STP cm^3/30 min. (Spec: 0.8 STP cm^3/30 min).	Check valve removed and replaced.

Table 25 (contd)

Model	Problem	Solution/Remarks
PTM (contd)	Service valves FA and OA leakage 0.78 STP cm^3/5 min and 0.8 STP cm^3/5 min with GN_2, respectively (spec: 0.5 STP cm^3/5 min).	Valve relapped and relubricated. Procedures changed to reduce closing torque from 282.4 cm-N (25 in.-lb) to 169.5 cm-N (15 in.-lb) for all prelaunch testing.
	Service valve OB leakage 181 STP cm^3/5 min. with GN_2 (spec: 0.5 STP cm^3/5 min).	No corrective action taken since valve leakage was acceptable for fluid containment for subsequent testing. Repair prior to flight use required.
	Service valves OD and OC leakage 3.17×10^{-6} STP cm^3/s and 2.36×10^{-4} STP cm^3/s with helium, respectively (spec: 1×10^{-6} STP cm^3/s).	Valve seat and ball/screw assembly cleaned and relubricated.
	Service valve FC leakage 2.52×10^{-6} STP cm^3/s with helium (spec: 1×10^{-6} STP cm^3/s).	Valve seat and screw/ball assembly cleaned and relubricated.
	Service valve PB high leak rate after pressurization of pressurant tanks to 3448×10^3 N/m^2 (500 psia).	Severe degradation of seat of valve -- nonreparable PCA replaced.
Flight I	During regulator functional test, pressure in test volume upstream of regulator decayed rapidly, indicating potential flight regulator internal leak.	Fill valve assembly engagement to service valve improper causing leakage at fill valve/service valve interface. Reattachment and retest successful.
	Service valves OC and FC leakage 2.5×10^{-6} and 8×10^{-6} STP cm^3/s with helium, respectively (spec: 1×10^{-6} STP cm^3/s).	Ceramic balls replaced; valves cleaned and relubricated with Krytox 143 AB.

Table 25 (contd)

Model	Problem	Solution/Remarks
Flight I (contd)	Service valves PA and PB leakage, 2.75×10^{-4} and 3.5×10^{-4} STP cm^3/s with helium, respectively (spec: 1×10^{-6} STP cm^3/s).	Ceramic balls replaced; valves cleaned and relubricated with Krytox 143 AB.
	Oxidizer and fuel bladder leakage 5.8×10^{-2} STP cm^3/s and 6.0×10^{-2} STP cm^3/s, respectively (spec: 5.0×10^{-2} STP cm^3/s).	Specification leak rate increased from 5.0×10^{-2} to 6.6×10^{-2} STP cm^3/s. Bladder material change to codispersion laminate -- higher permeability.
	Fuel line pressure transducer indicated dropouts at and above 2034×10^3 N/m^2 (295 psia).	Use as is -- noise observed during transient condition. Flight data taken under steady-state conditions.
	PC and RO transducer dropouts at and above 2310×10^3 N/m^2 (335 psia).	Use as is -- noise observed during transient condition. Flight data taken under steady-state conditions.
Flight II	Check valve leakage during attempts to Freon flush PC and RO (S/N 004).	Check valve removed and replaced.
	Service valve FC leakage 1.53×10^{-5} STP cm^3/s (spec: 1×10^{-6} STP cm^3/s).	Replaced ceramic ball with smoother surface ball; relapped valve seat.
	Service valve PA leakage 8.54×10^{-4} STP cm^3/s (spec: 1×10^{-6} STP cm^3/s).	Replaced ceramic ball with smoother surface ball; relapped valve seat.
	Line pressure transducers noisy at and above 1655×10^3 N/m^2 (240 psia) (oxidizer) and 2172×10^3 N/m^2 (315 psia) (fuel), respectively.	Use as is -- noise observed during transient condition. Flight data taken under steady-state conditions.
	Apparent leakage of fuel and oxidizer relief valve burst disks 3.3×10^{-4} STP cm^3 GN_2/s and 2.1×10^{-4} STP cm^3 GN_2/s (spec: zero).	Determined to be a function of test technique -- temperature of hardware, venting rate of gas downstream of burst disk. No leakage problem.

Table 25 (contd)

Model	Problem	Solution/Remarks
Flight II (contd)	Apparent leakage of oxidizer check valve, 7.2×10^{-4} STP cm^3 GN_2/s (spec: 2.2×10^{-4} STP $cm^3 GN_2/s$). After 16-h period, retest indicated zero leakage.	Allowable leakage spec. increased to 4.8×10^{-4} STP cm^3 GN_2/s. Indicated leakage apparently a function of hardware and gas temperature in check valve -- regulator cavity. No real leakage existed.
	Pressure regulator outlet fitting braze joint X-ray indicated poor bond.	Regulator and filter removed and replaced.
	Oxidizer check valve leakage after replacement of regulator and filter 1.0×10^{-2} STP cm^3/s (spec: 4.8×10^{-4} STP cm^3/s).	Removed and replaced check valve. Probable cause -- contamination generated during replacement of upstream components.
	Bladder leak rates of oxidizer and fuel PIA's were 5.5×10^{-2} STP cm^3/s and 5.1×10^{-2} STP cm^3/s (spec: 5.0×10^{-2} STP cm^3/s).	Specification changed to 6.6×10^{-2} STP cm^3/s maximum -- new bladder material more permeable.
	Service valves OA, OC, and PB leakage above allowable (spec: 1×10^{-6} STP cm^3/s).	New, smoother ceramic balls installed in place of originals.
	Oxidizer propellant tank pressure transducer noisy.	Removed and replaced.

Table 26. Mariner Mars 1971 Propulsion Support Equipment

Operation	Support Equipment
Propellant tank assembly	Shipping container Propellant tank handling fixture Lifting fixture propellant tank
Thrust plate assembly	Engine/propulsion valve installation fixture Lifting fixture
Propulsion system buildup	Handling fixture Hoisting fixture Workstand Low-level positioner Braze joint X-ray unit
Proof, leak, and functional tests	Pneumatic control console Auxiliary pneumatic control console CEC helium leak detector Engine solenoid valve actuation console Transducer monitoring console Engine injector scanner
Fuel and oxidizer solvent loading	Fuel flush and oxidizer flush carts Weighing scale Solvent service carts
PSS vacuum chamber leak test	Mobile pressurization console High-pressure gas trailer Differential-pressure control panel Transducer monitoring console Transporter
Launch vibration test	Module transfer ring
Spacecraft systems test	Engine alignment fixture High pressure gas trailer Propulsion simulator
Propellant load at Edwards Test Station	Fuel and oxidizer service trailers Transducer monitoring console Stationary platform scale Pneumatic control console Fuel and oxidizer flush carts Rotation fixture Propellant tank X-ray unit

Table 26 (contd)

Operation	Support Equipment
Propellant loading at AFETR	Fuel and oxidizer service trailers Platform scale Pneumatic control console Fuel and oxidizer flush carts Transducer monitoring console Data logger Toxic vapor detector Propellant tank X-ray unit SCAPE suits

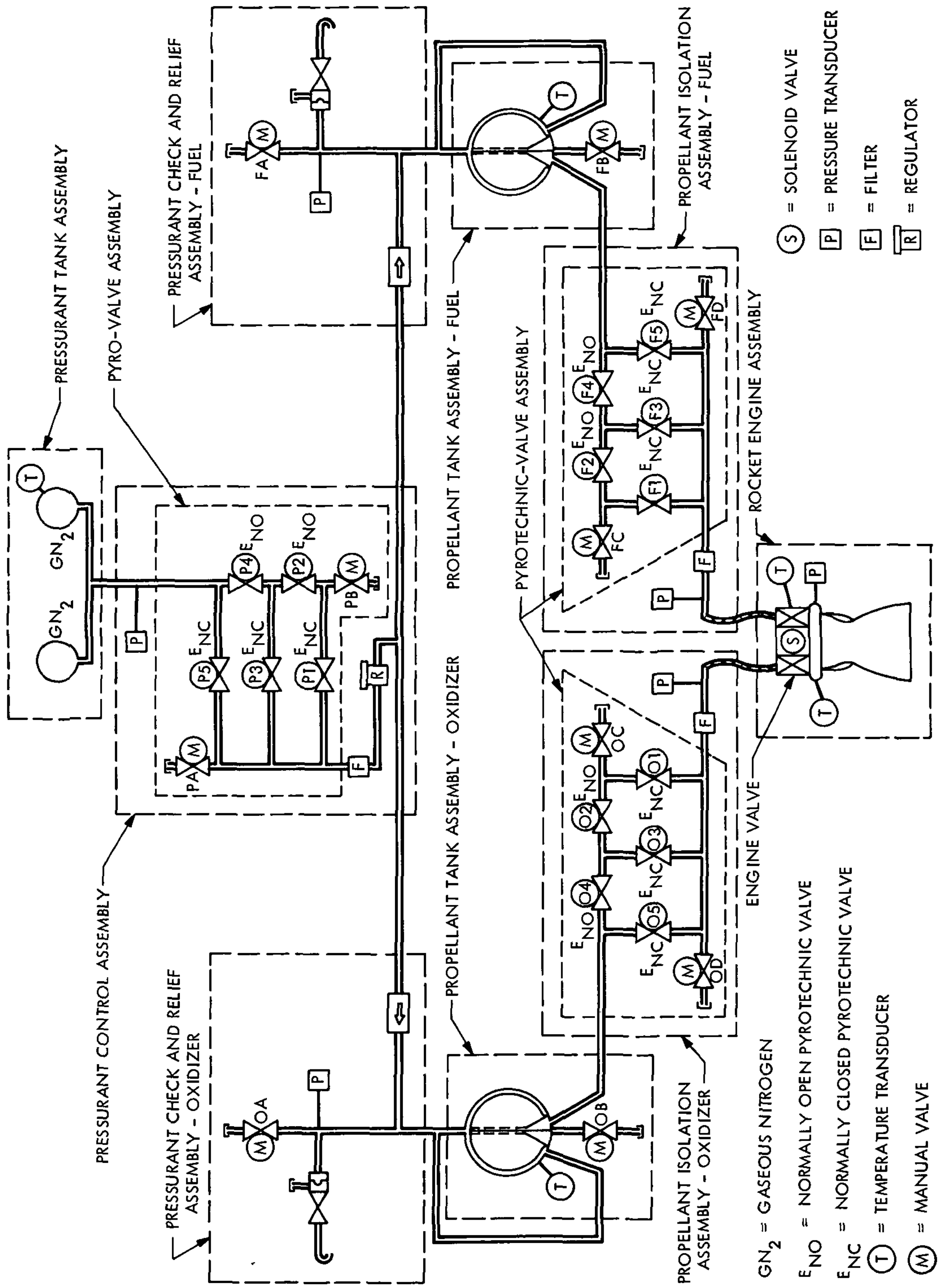


Fig. 1. Mariner Mars 1971 propulsion subsystem

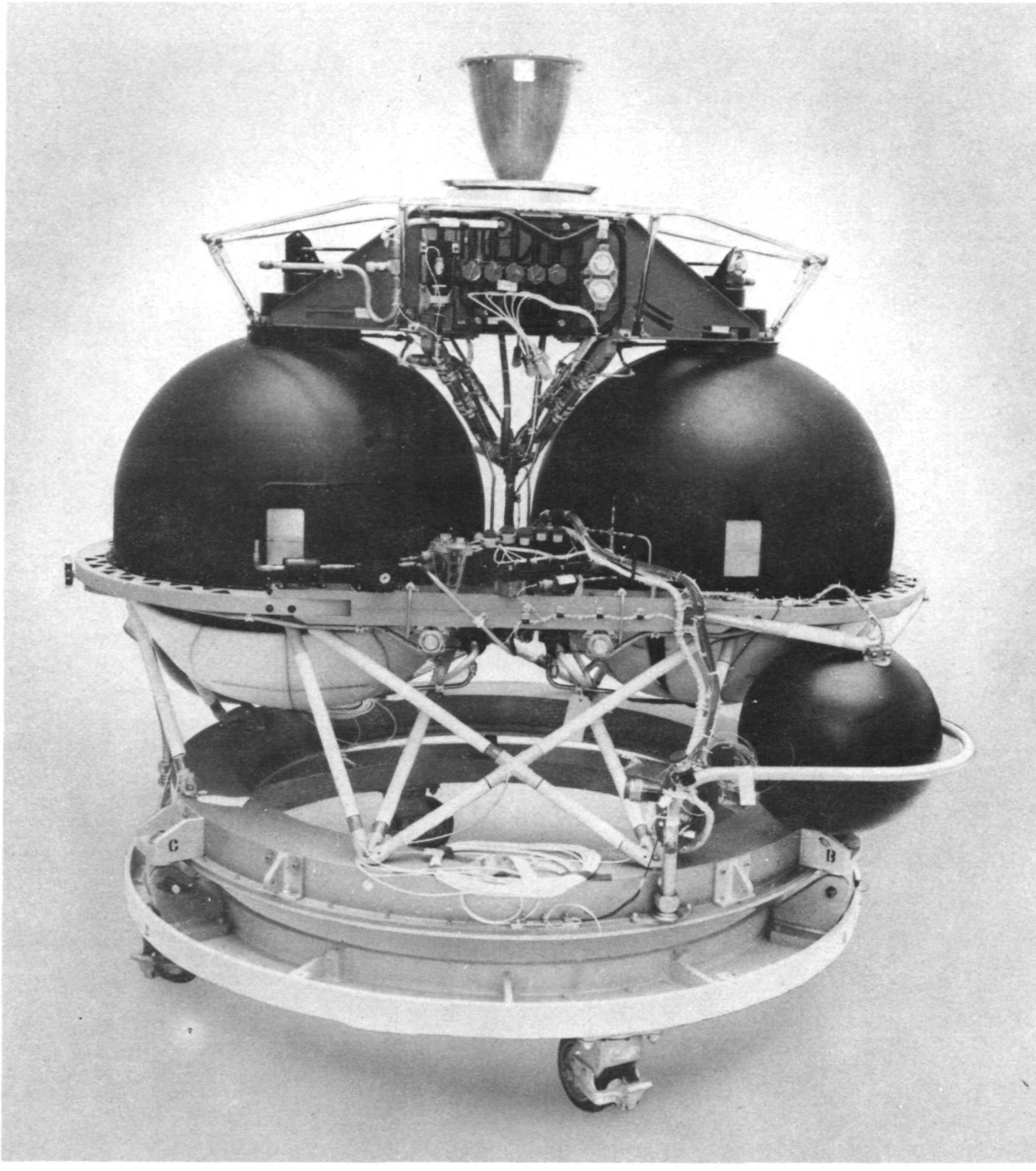


Fig. 2. Engineering test model after assembly

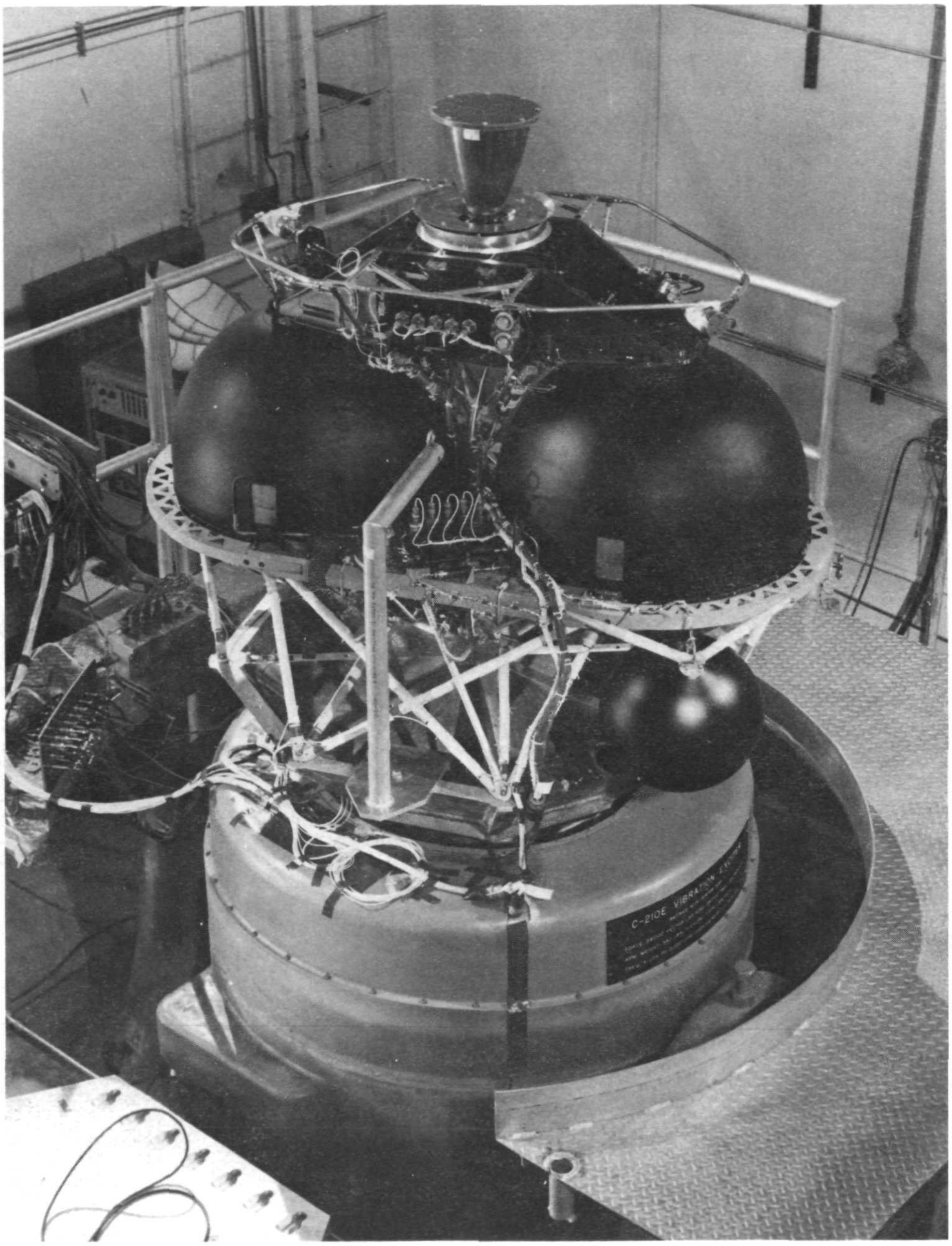


Fig. 3. Engineering test model Z-axis vibration

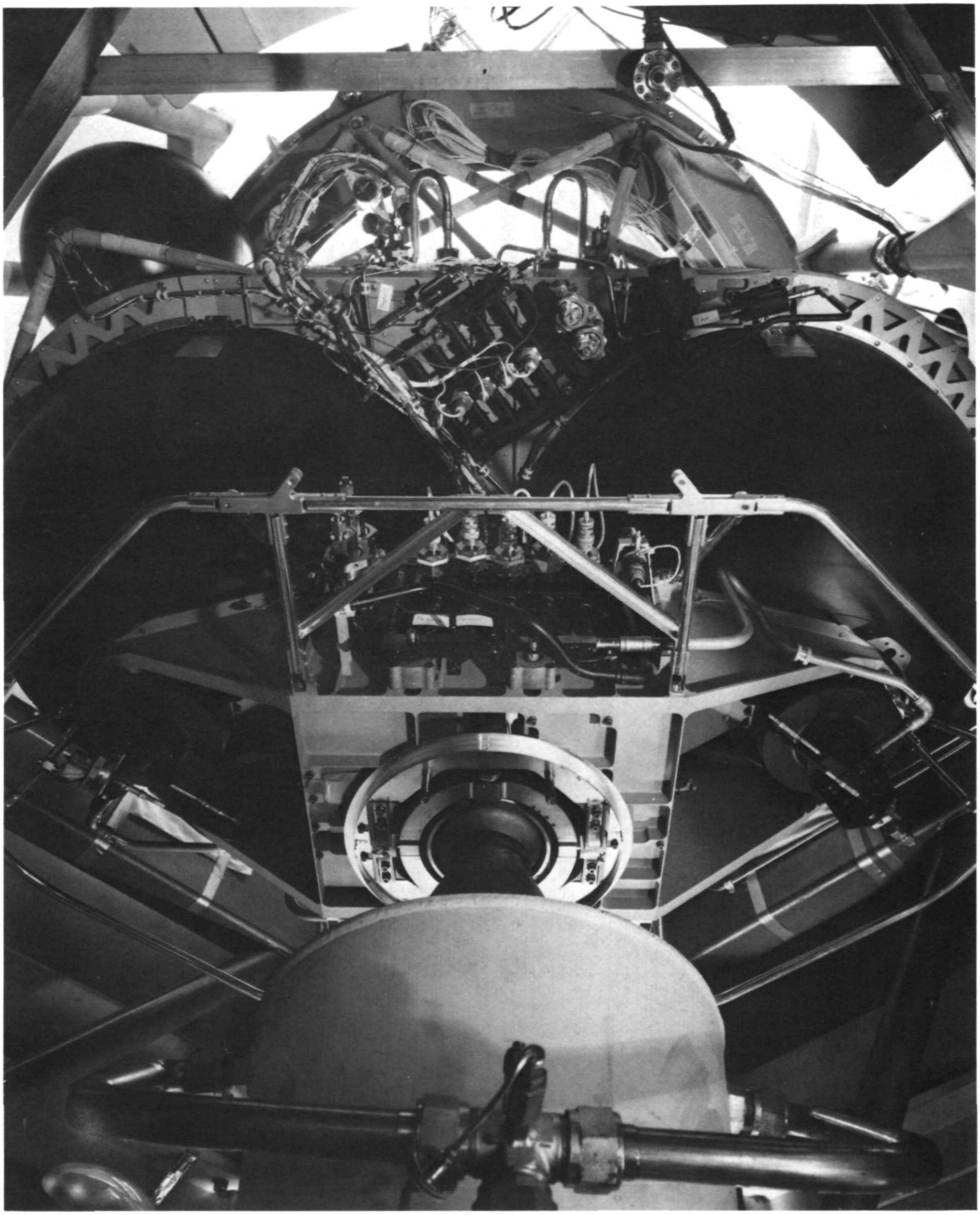
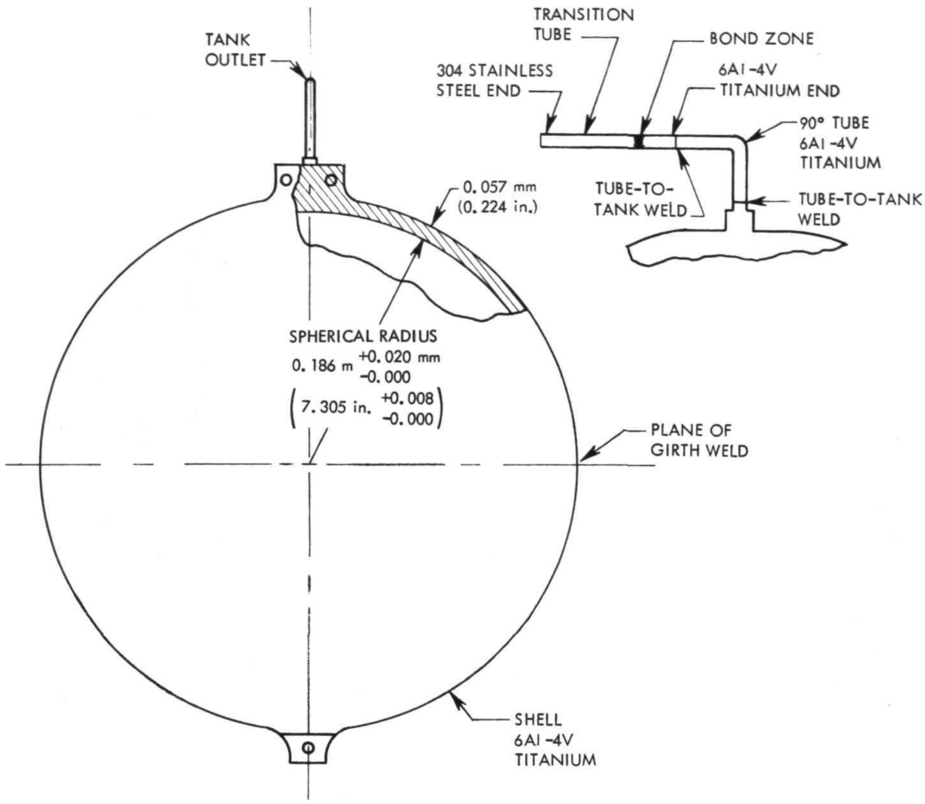
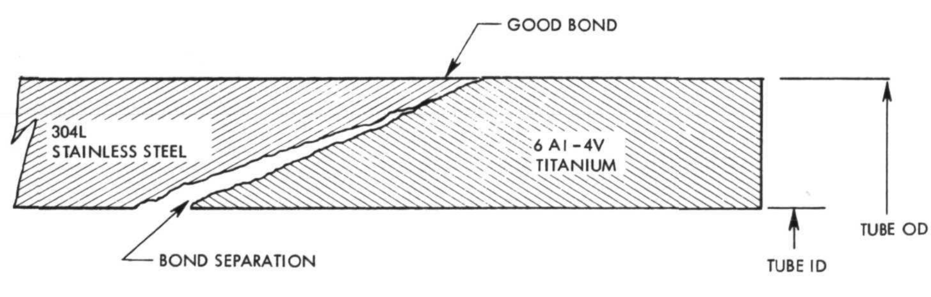


Fig. 4. Type approval propulsion subsystem mounted in vacuum chamber with thermal blanket removed, view from below the propulsion subsystem

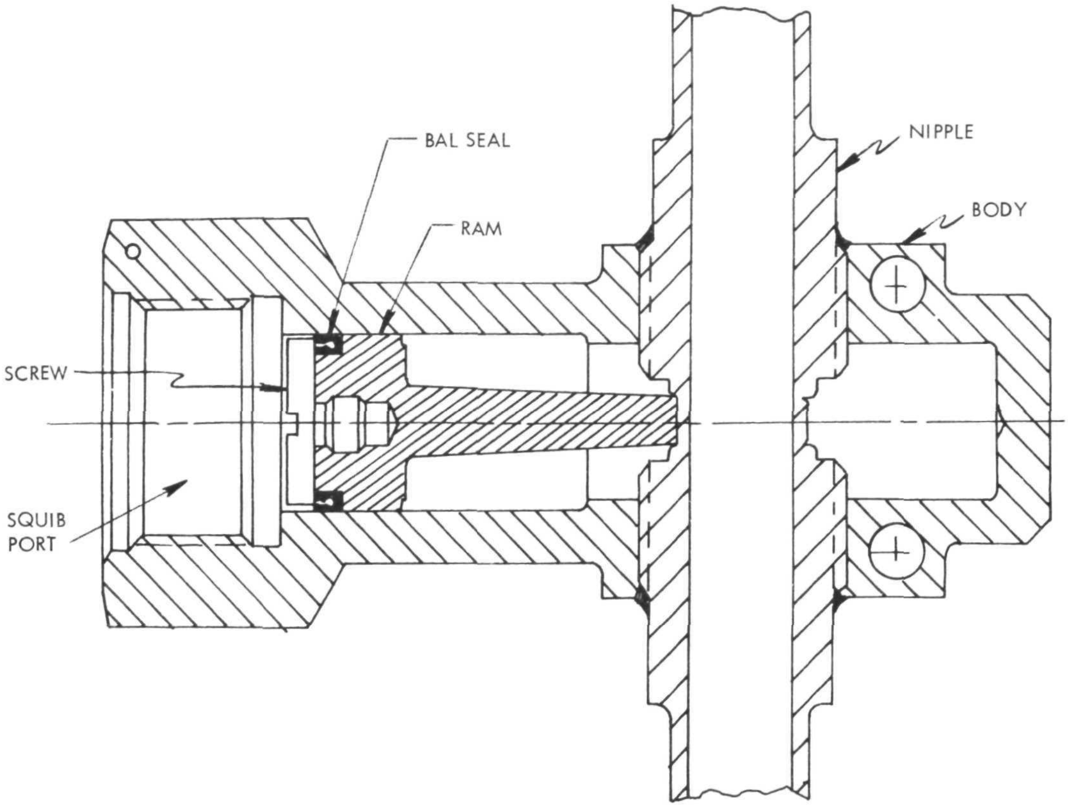


A) PRESSURANT TANK

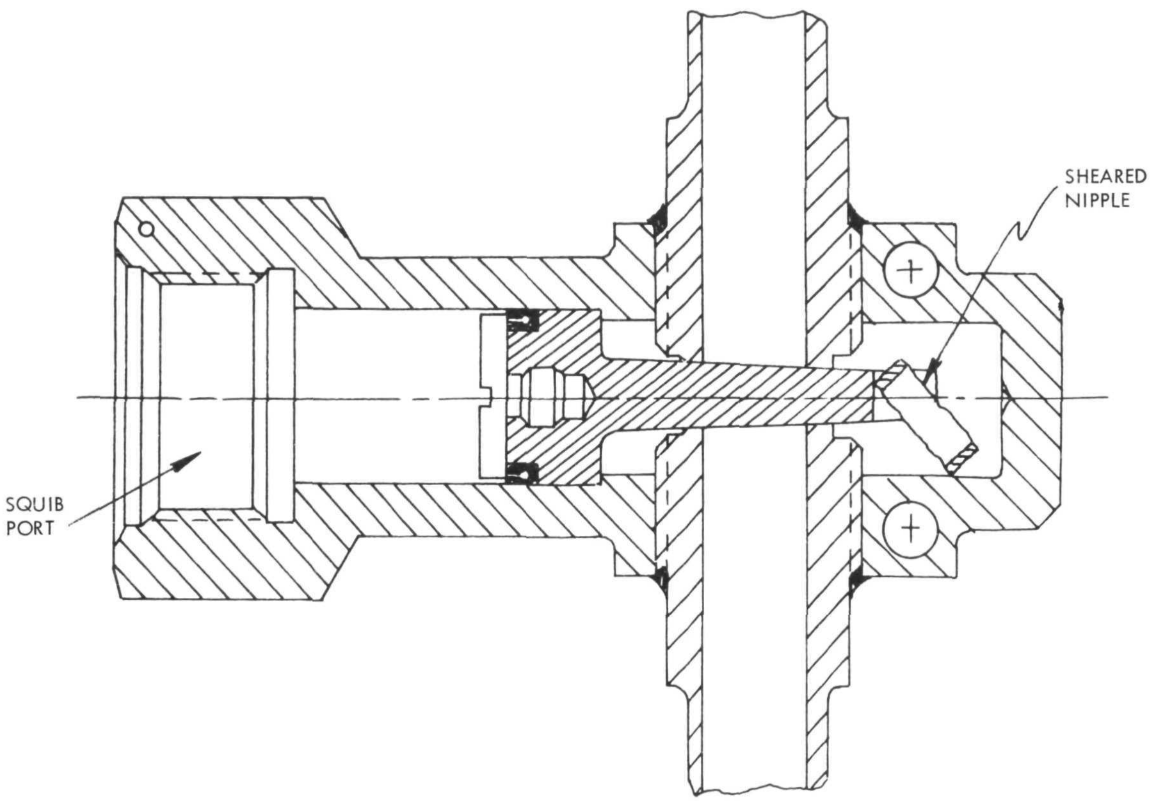


B) TRANSITION TUBE WALL CROSS SECTION SCHEMATIC, SHOWING BOND SEPARATION

Fig. 5. Pressurant tank and transition tube wall cross section



A) BEFORE ACTUATION



B) AFTER ACTUATION

Fig. 6. The normally open pyrotechnic valve

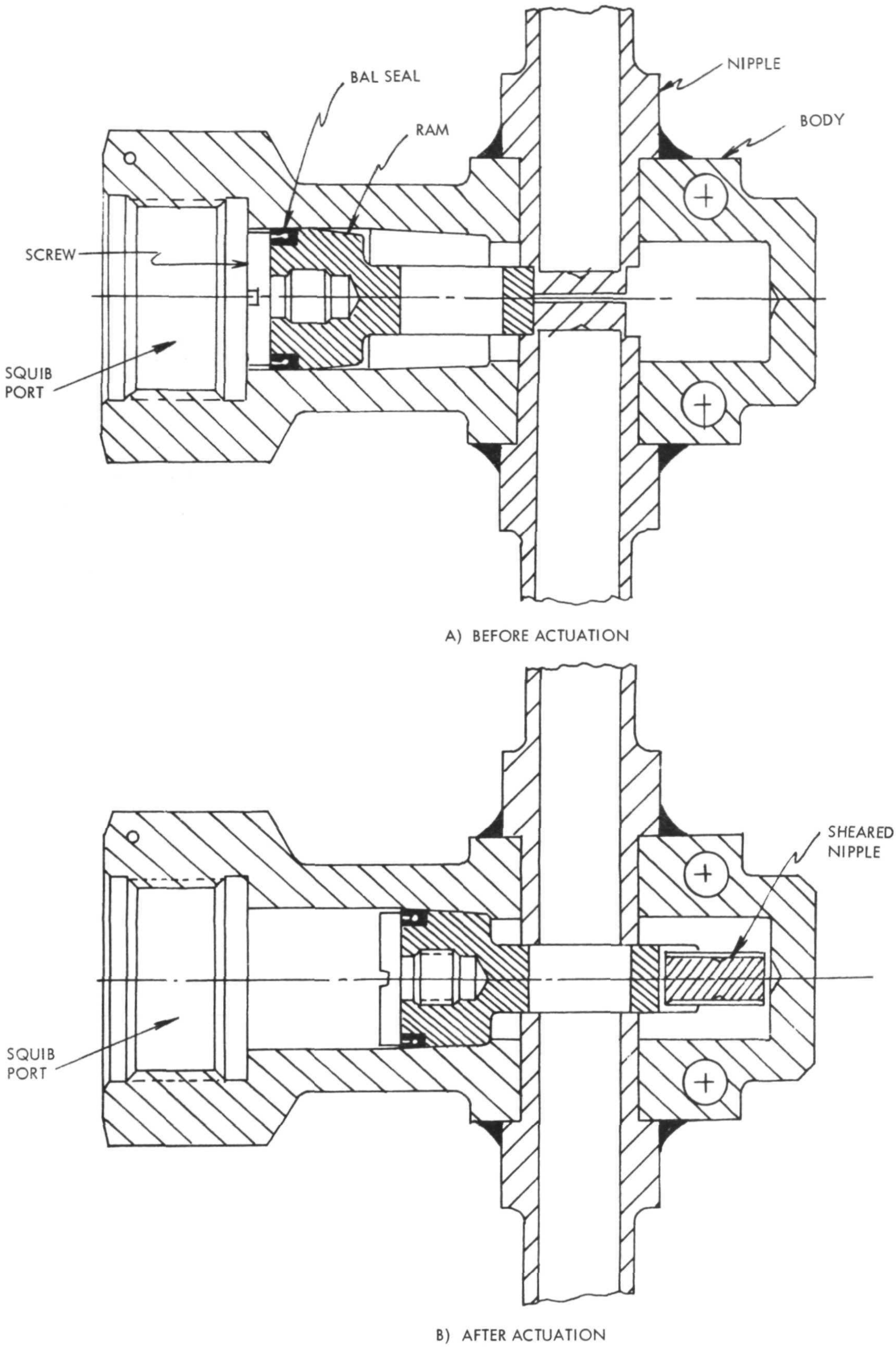


Fig. 7. The normally closed pyrotechnic valve

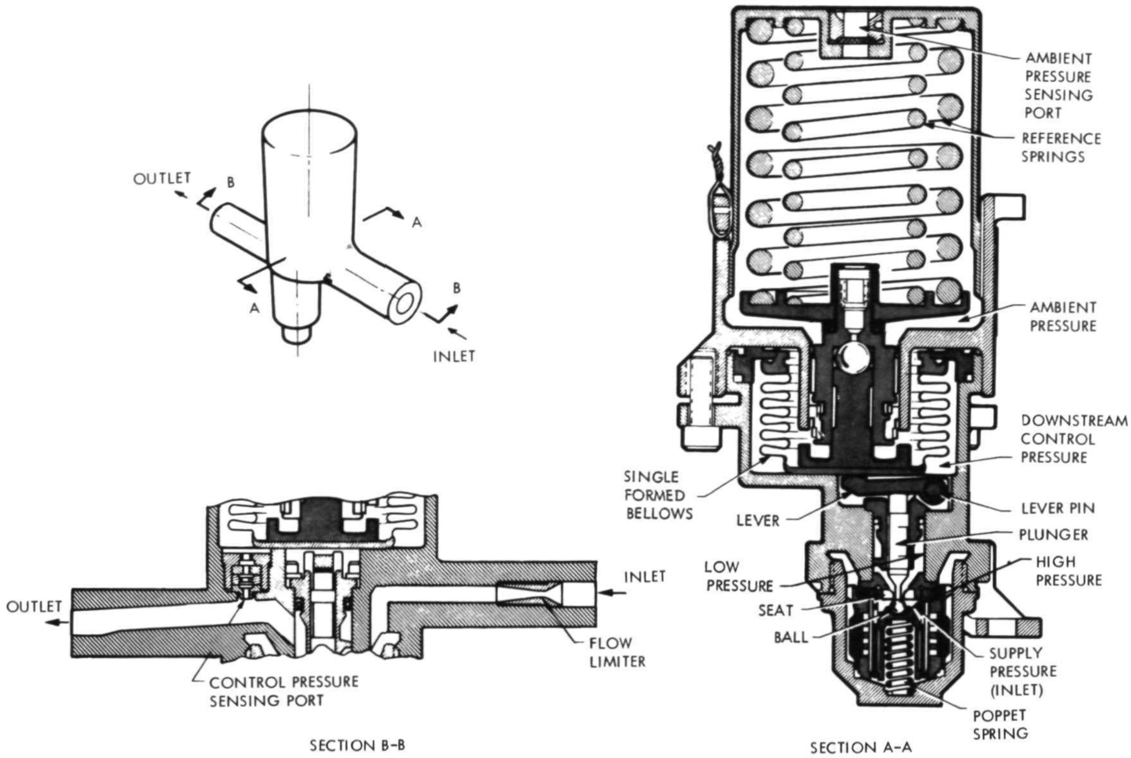


Fig. 8. Mariner Mars 1971 pressure regulator, sectional views

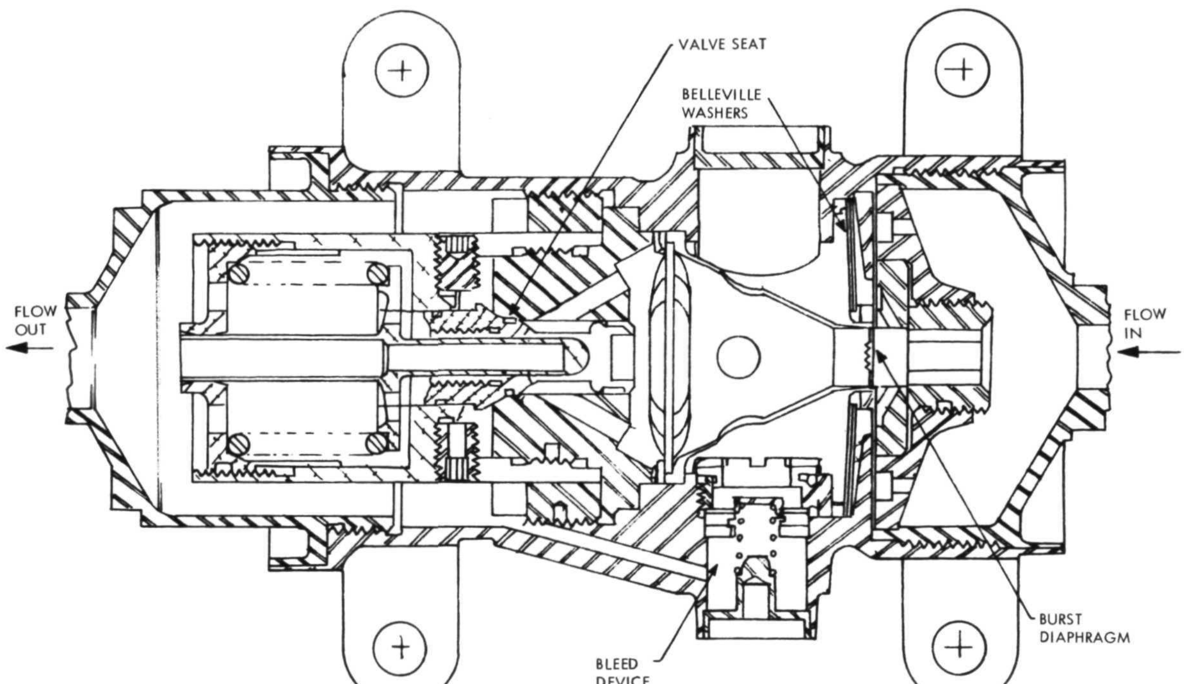


Fig. 9. Pressurant relief valve

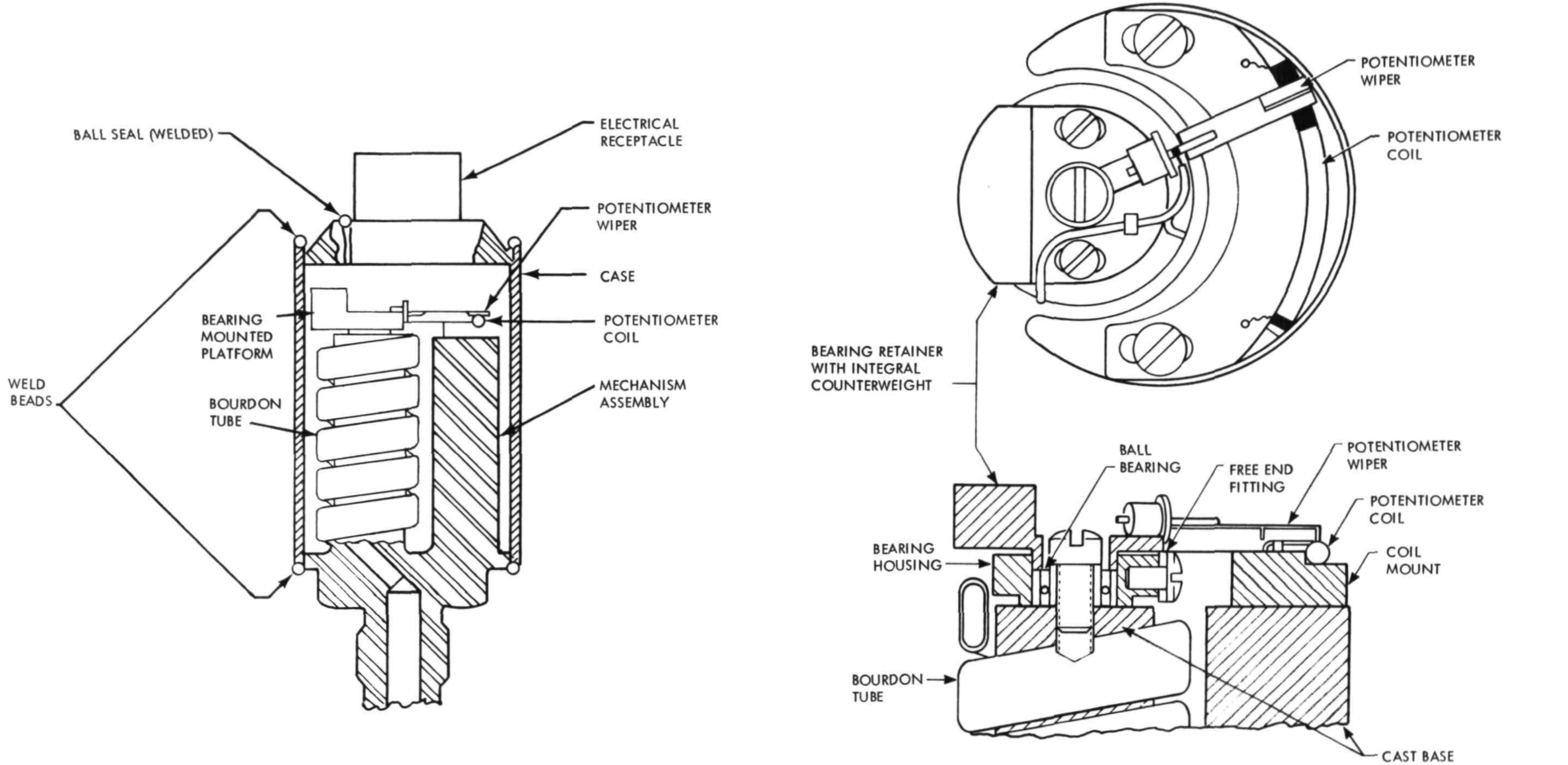


Fig. 10. High-pressure transducer, cross sections

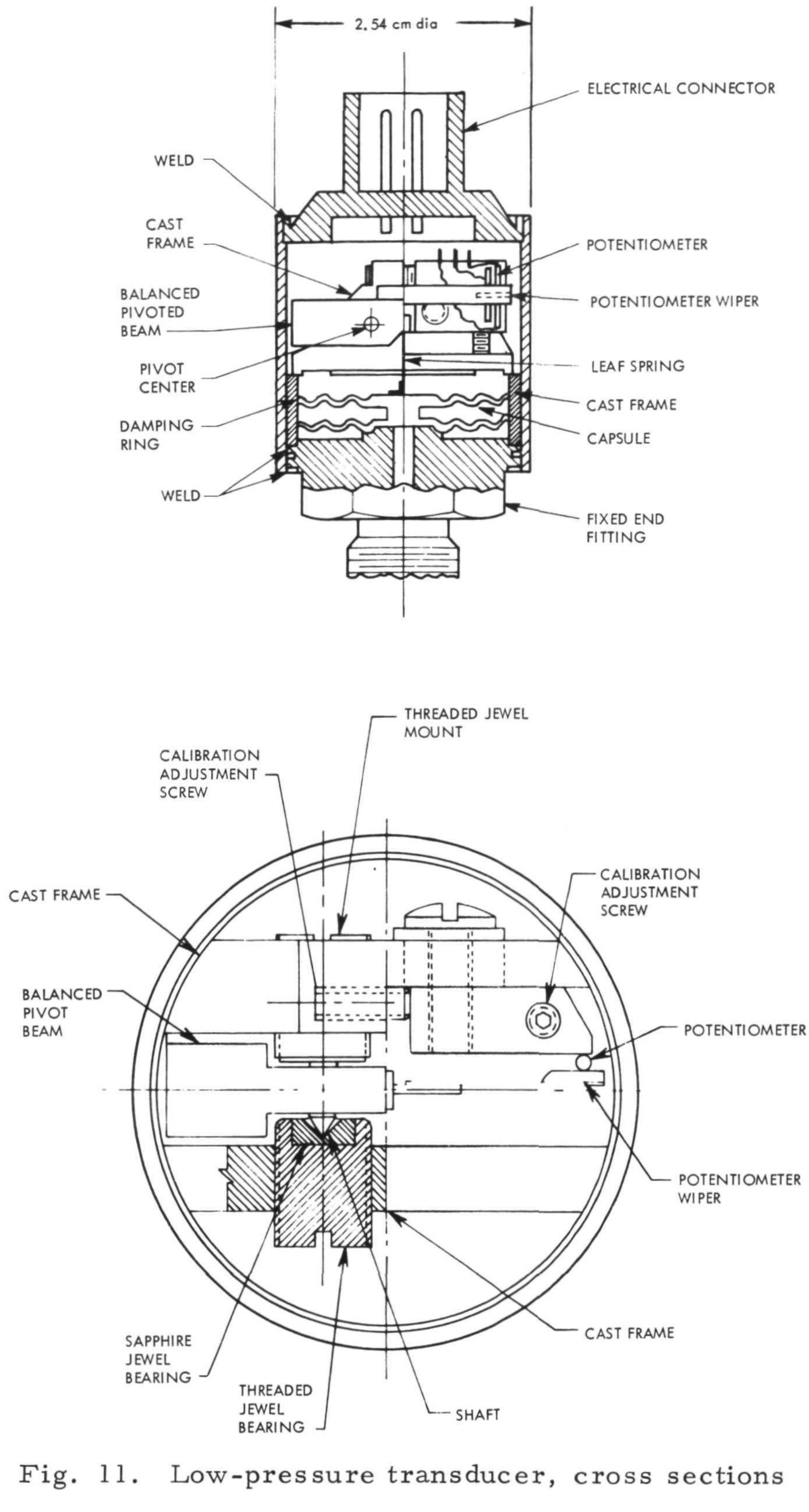


Fig. 11. Low-pressure transducer, cross sections

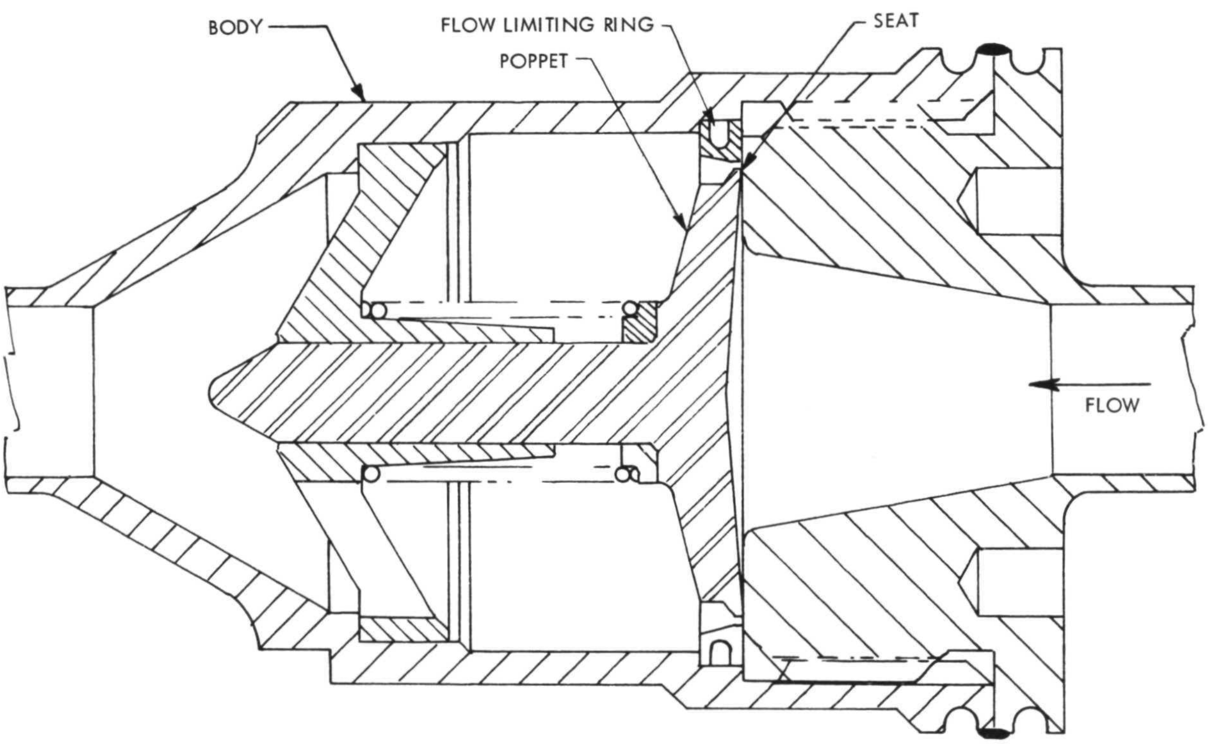


Fig. 12. Check valve, cross section

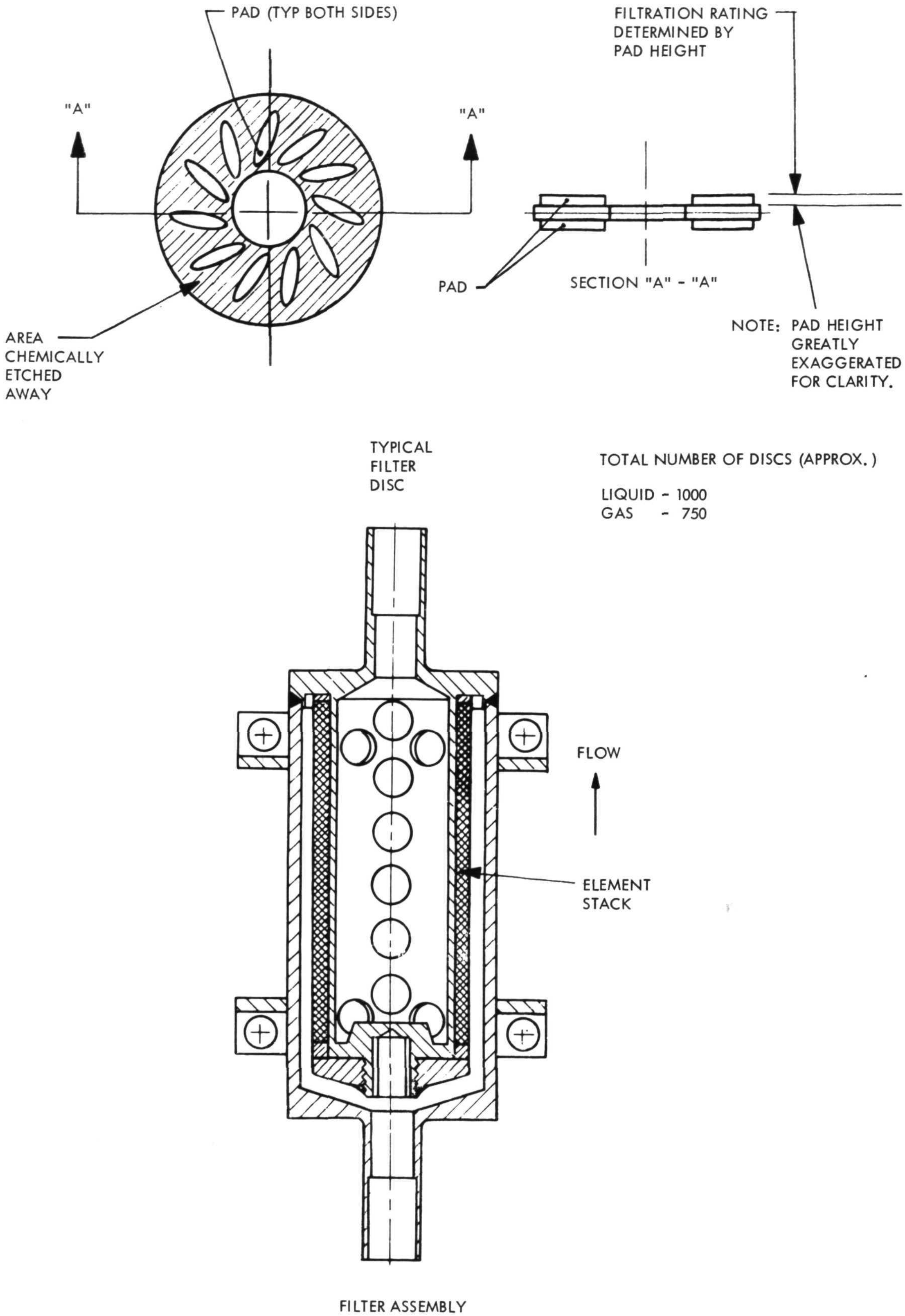


Fig. 13. Filter assembly cross section and disk detail

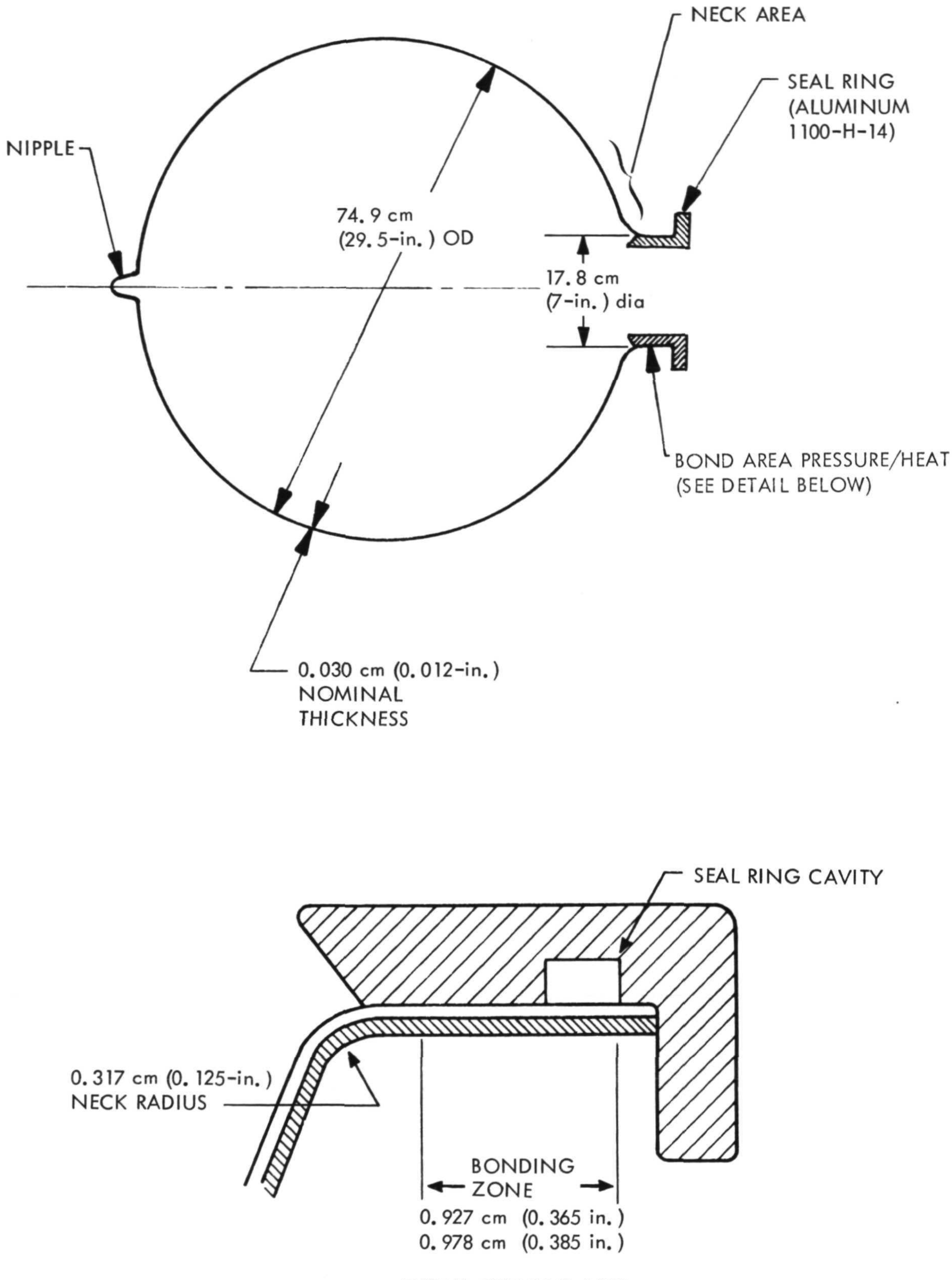


Fig. 14. Production codispersion bladder configuration

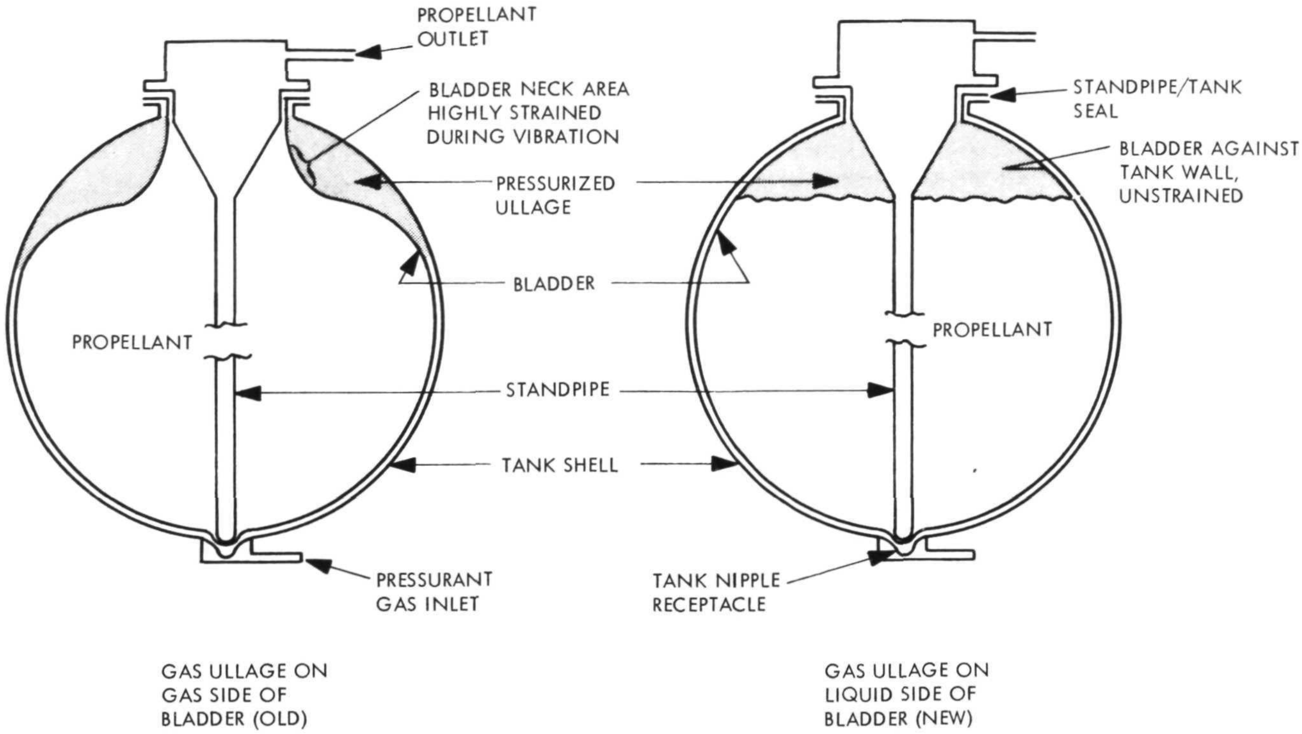


Fig. 15. Propellant tank gas ullage orientations

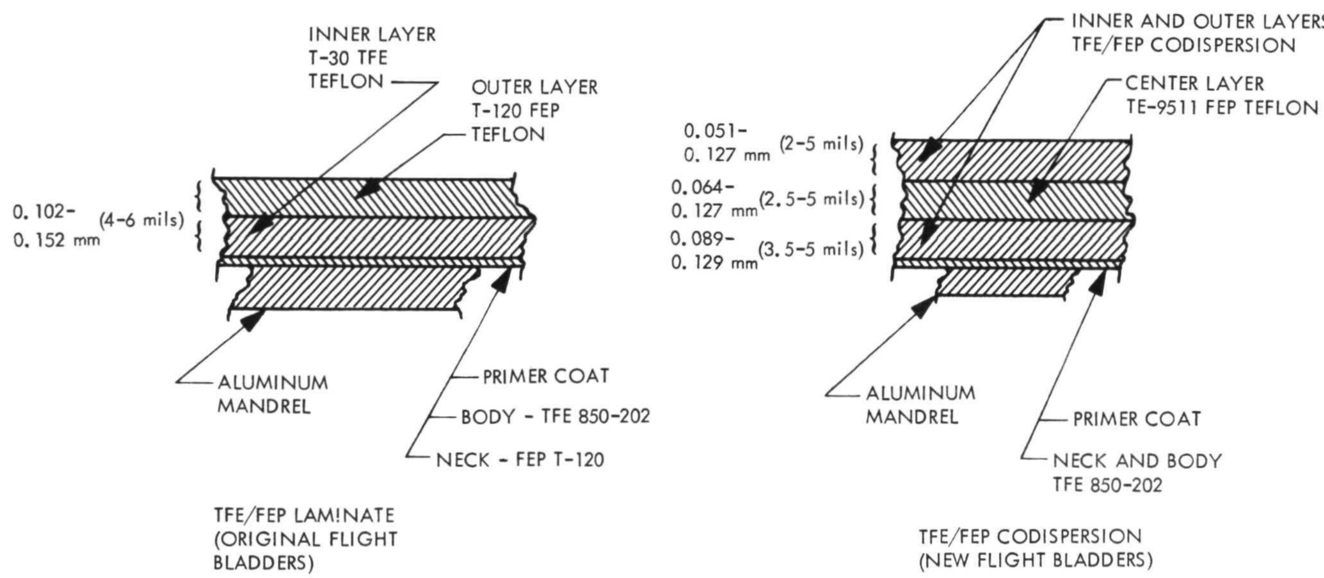


Fig. 16. Bladder membrane construction

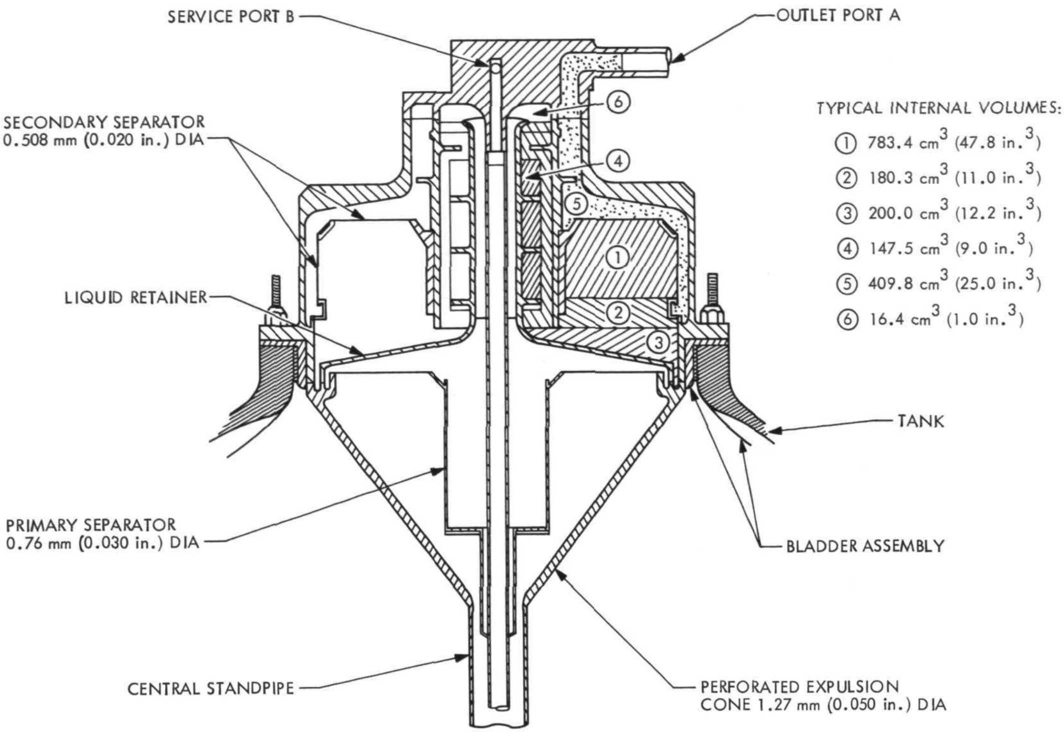


Fig. 17. Propellant standpipe and acquisition trap assembly

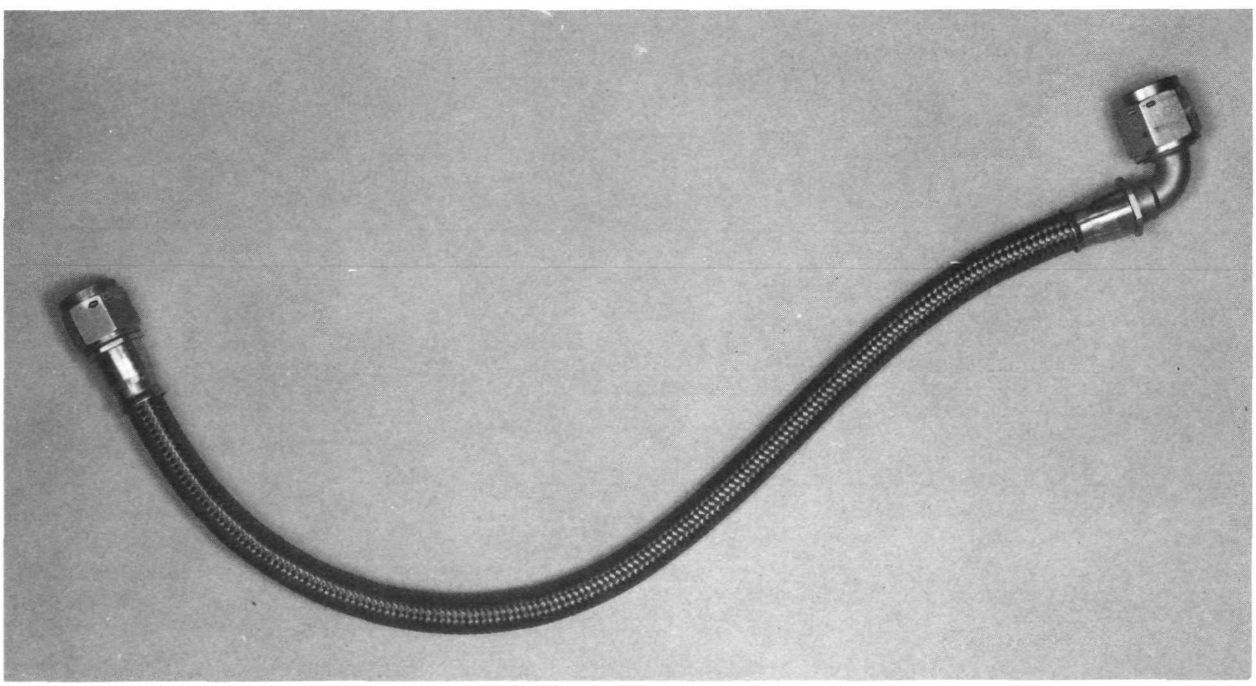


Fig. 18. Feedline hose assembly

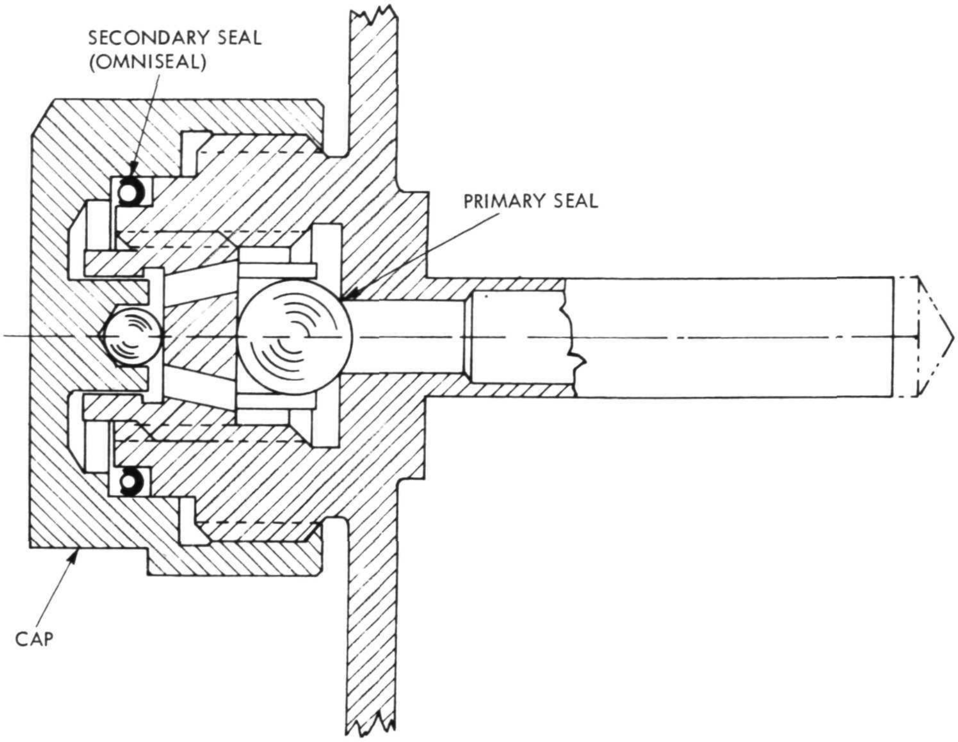


Fig. 19. Servicing valve assembly, airborne half, cross section

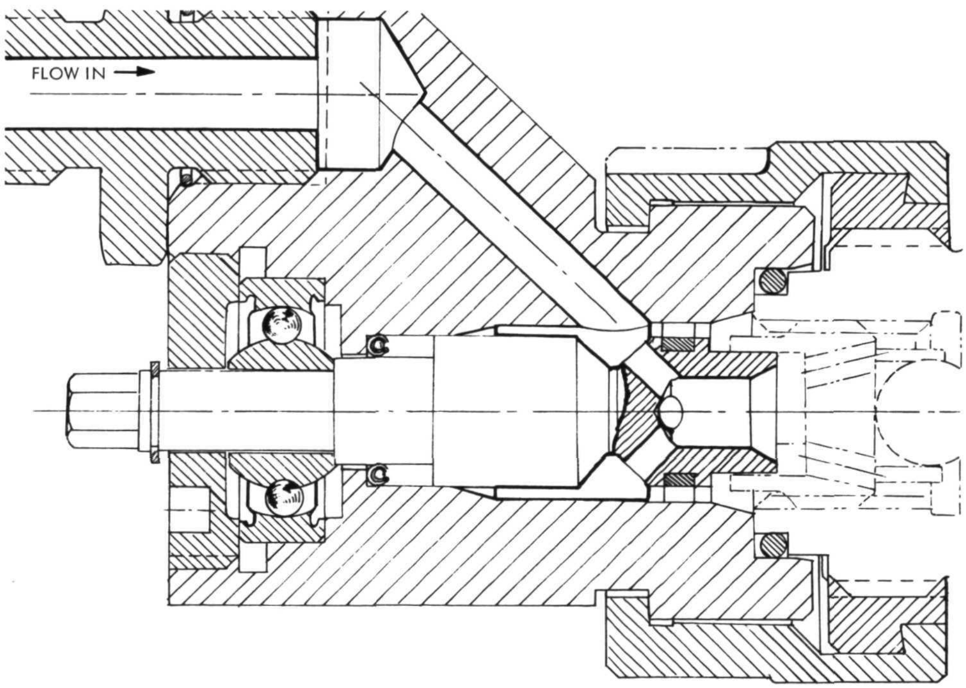


Fig. 20. Servicing valve assembly, ground half, cross section

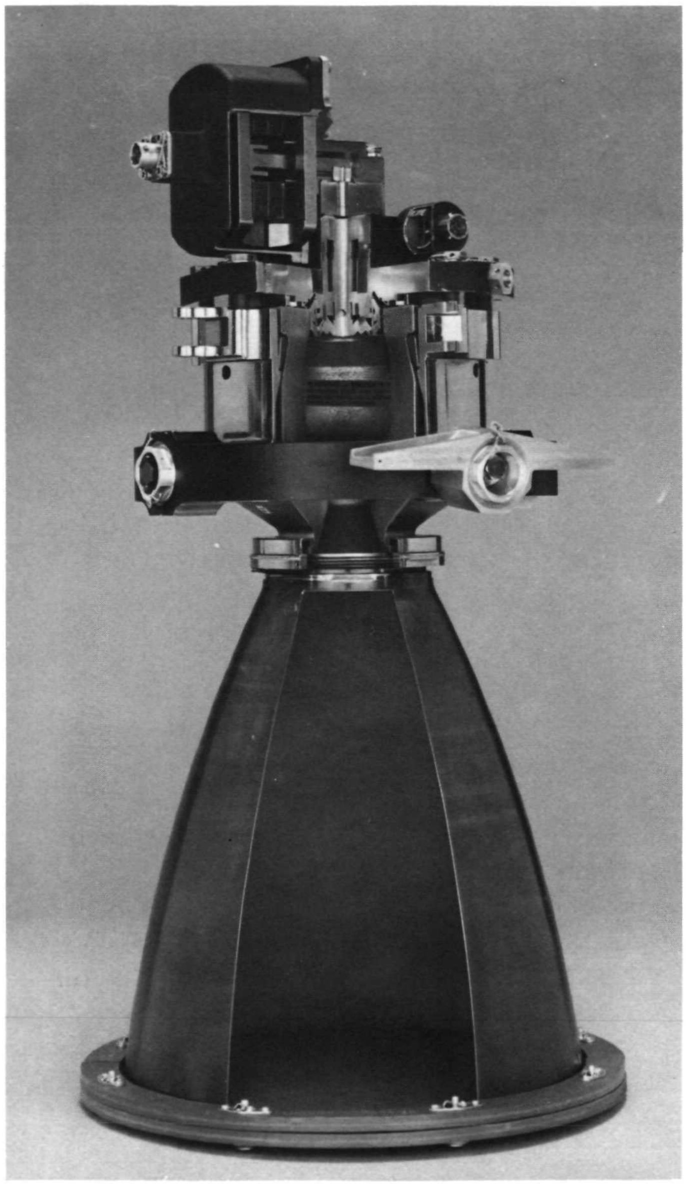


Fig. 21. Mariner Mars 1971 rocket engine assembly

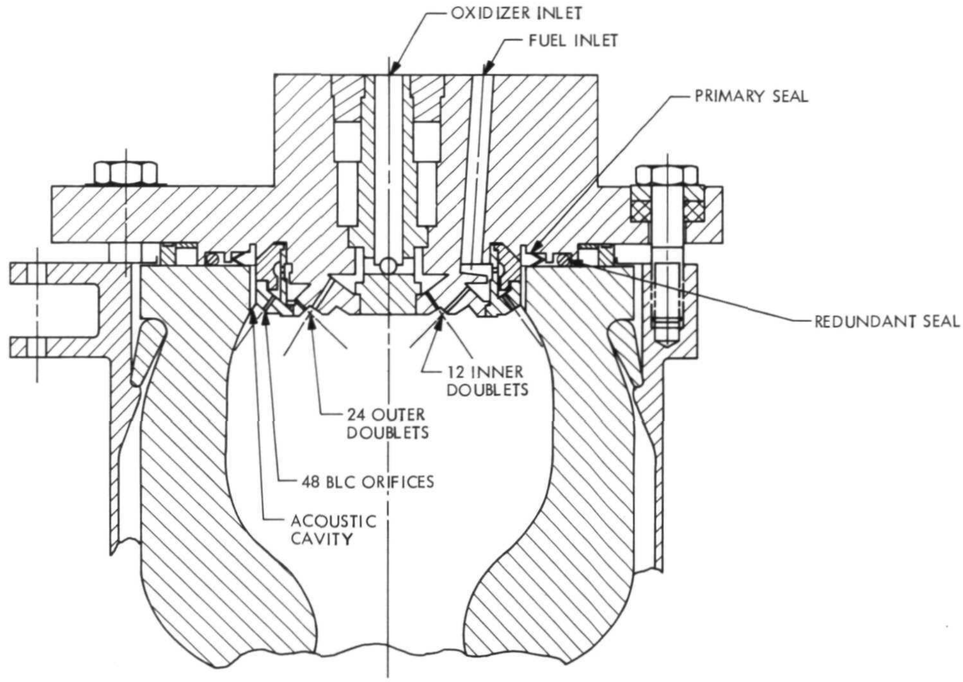


Fig. 22. Injector/thrust chamber interface, cross section

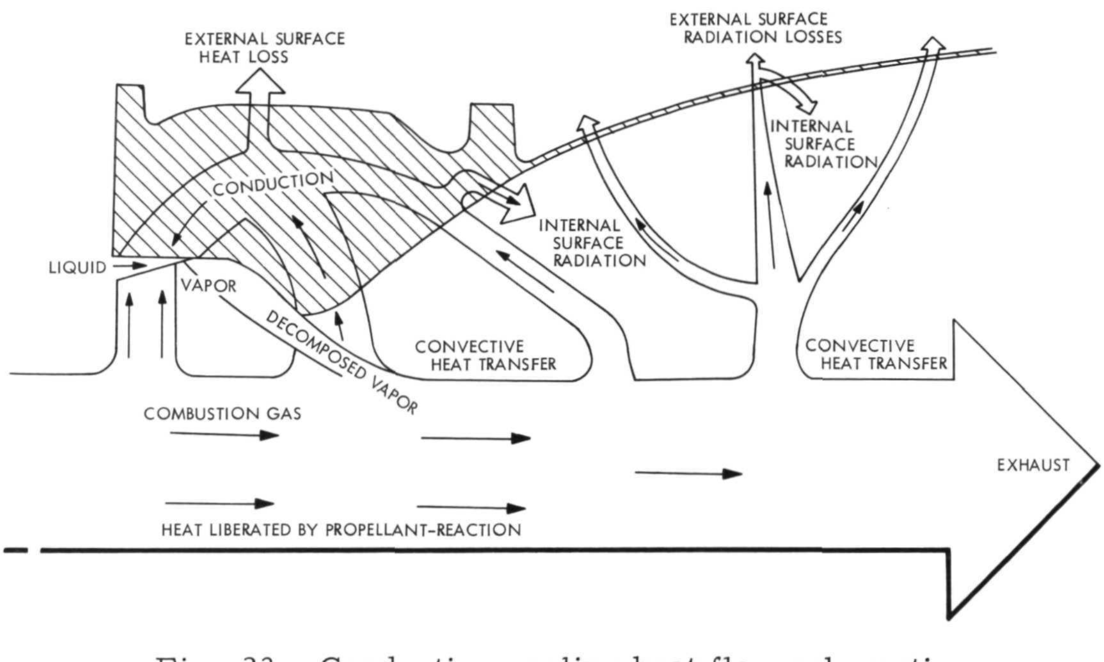


Fig. 23. Conduction cooling heat flow schematic

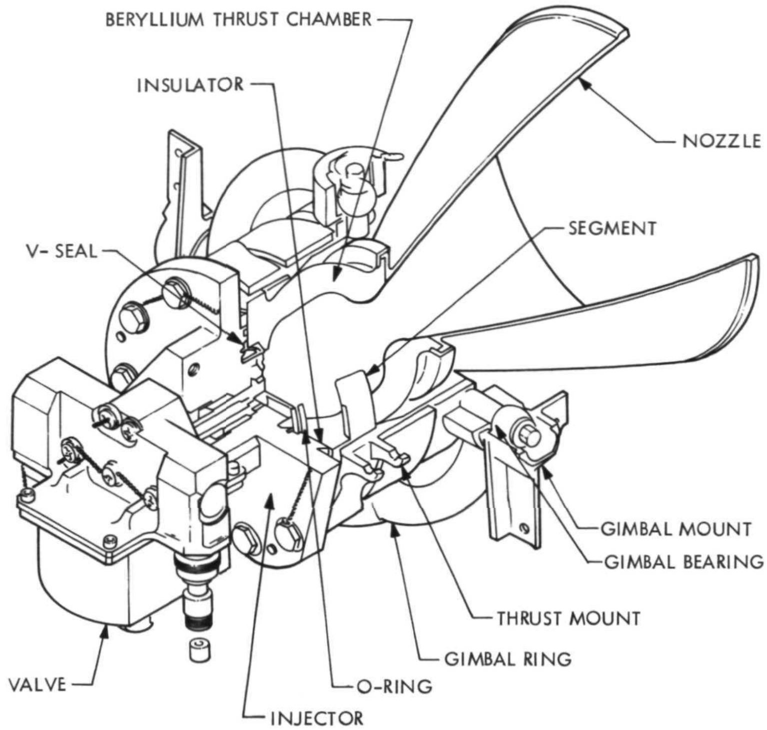


Fig. 24. Mariner Mars 1971 rocket engine assembly

Fig. 25. Rocket engine valve

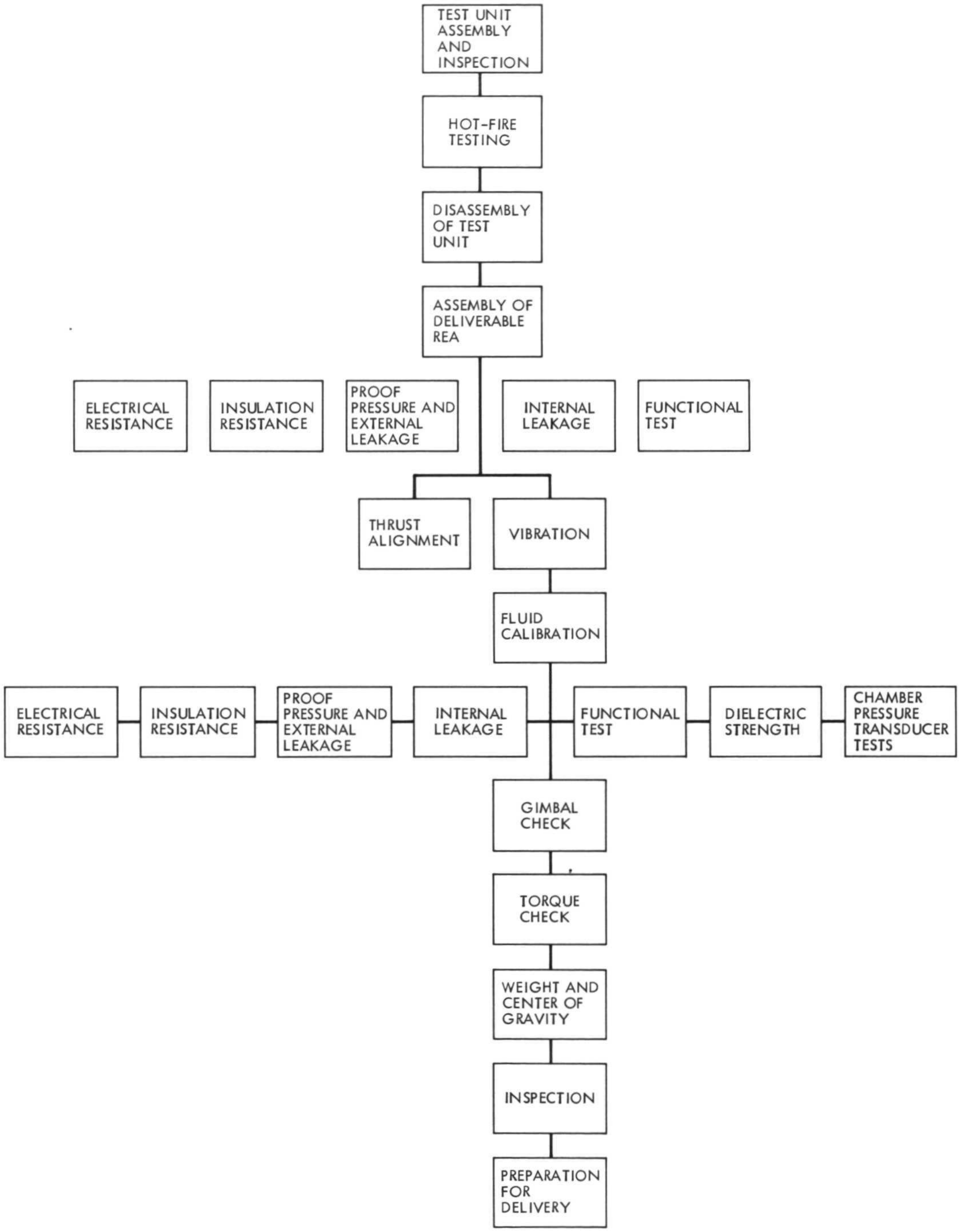


Fig. 26. Rocket engine assembly acceptance test sequence

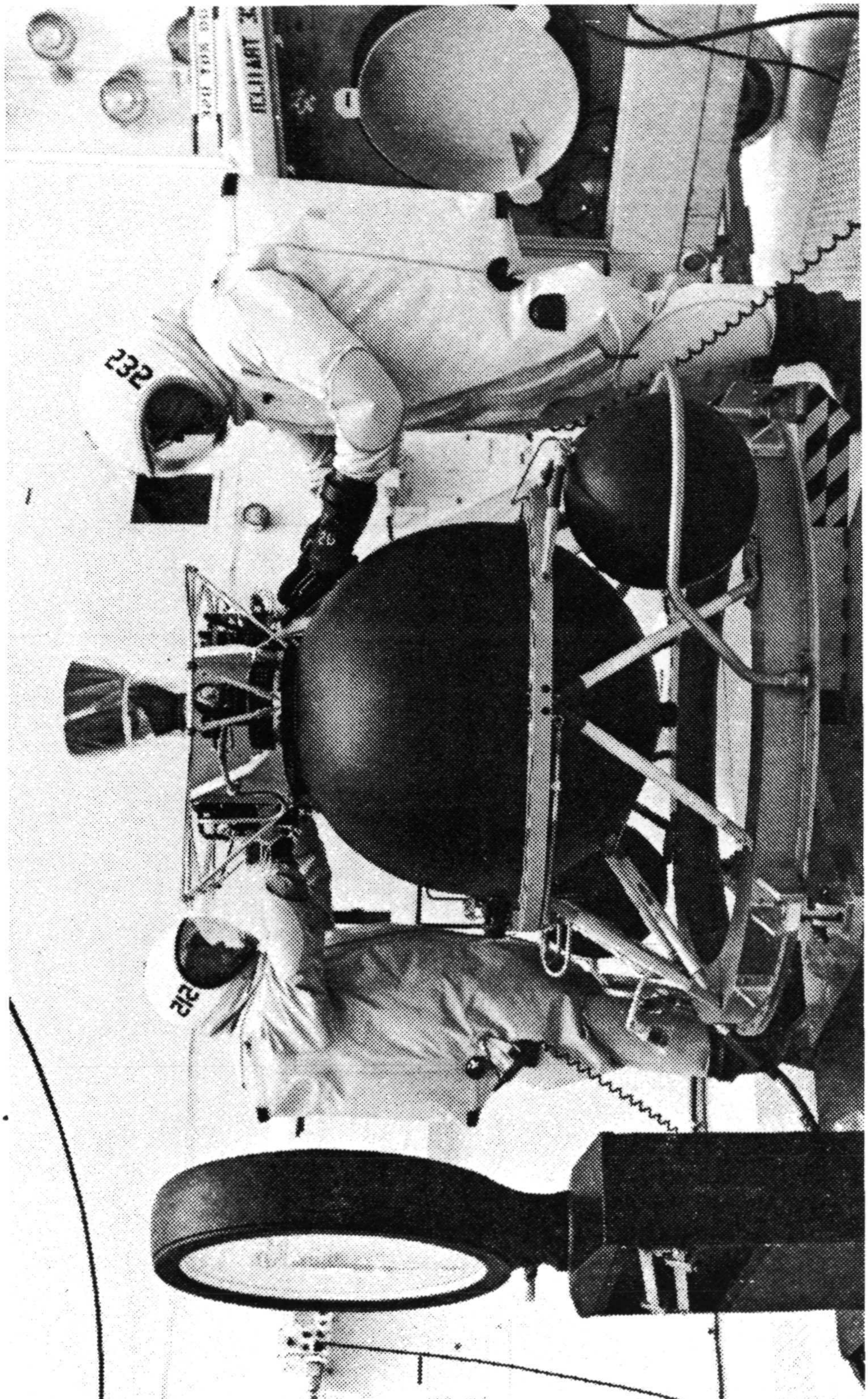


Fig. 27. Propellant loading operation at AFETR using SCAPE suits

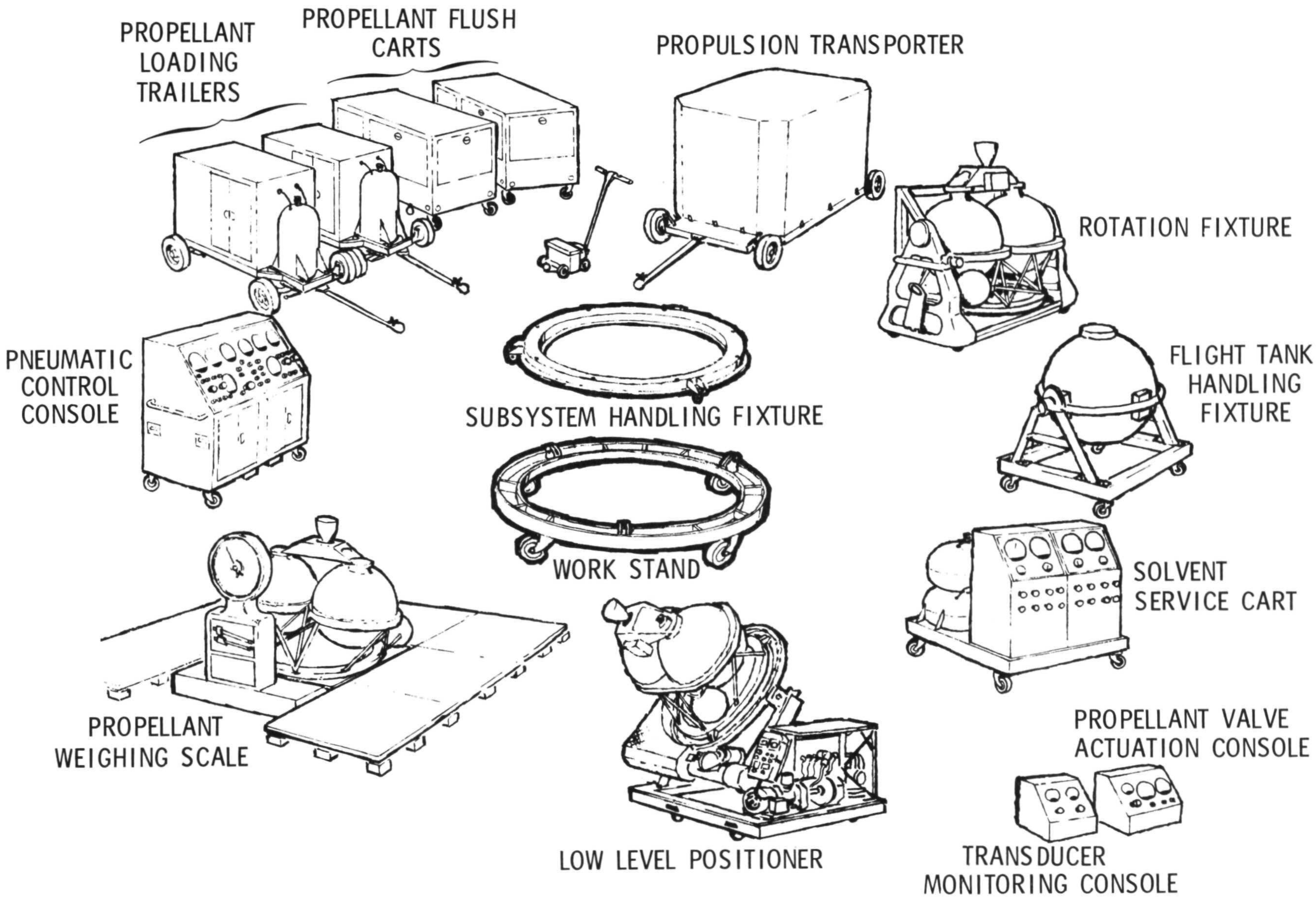


Fig. 28. Propulsion support equipment

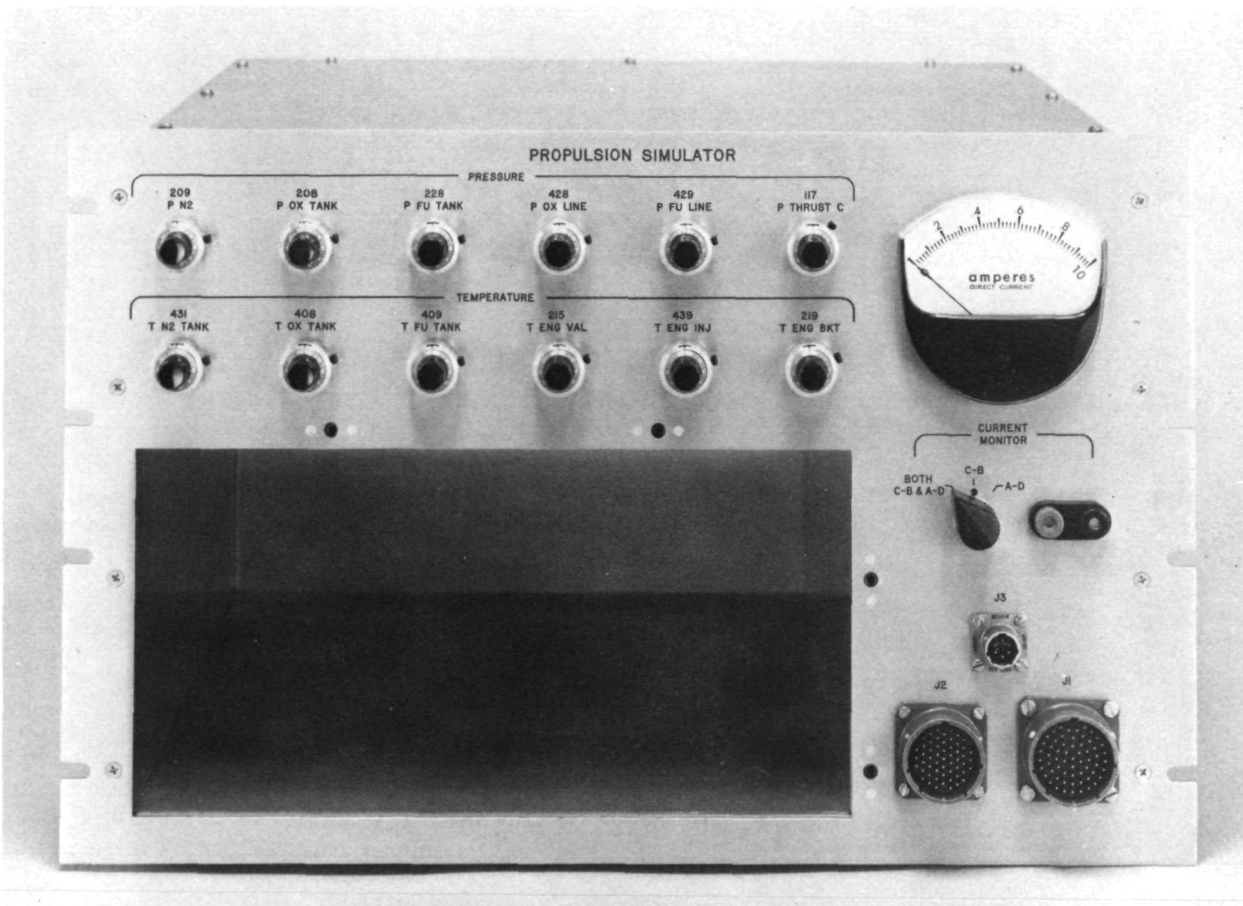


Fig. 29. Propulsion simulator

TECHNICAL REPORT STANDARD TITLE PAGE

1. Report No. 33-552	2. Government Accession No.	3. Recipient's Catalog No.

4. Title and Subtitle DEVELOPMENT AND TESTING OF THE PROPULSION SUBSYSTEM FOR THE MARINER MARS 1971 SPACECRAFT	5. Report Date August 1, 1972 6. Performing Organization Code
7. Author(s) Richard D. Cannova et al.	8. Performing Organization Report No.
9. Performing Organization Name and Address JET PROPULSION LABORATORY California Institute of Technology 4800 Oak Grove Drive Pasadena, California 91103	10. Work Unit No. 11. Contract or Grant No. NAS 7-100
12. Sponsoring Agency Name and Address NATIONAL AERONAUTICS AND SPACE ADMINISTRATION Washington, D.C. 20546	13. Type of Report and Period Covered Technical Memorandum 14. Sponsoring Agency Code

15. Supplementary Notes

16. Abstract

In November 1971, the Mariner 9 spacecraft was injected into Martian orbit by a 574-kg (1265 lb_m) propulsion system. Design of that system provided directed impulse, upon command, to accomplish in-transit trajectory corrections, an orbital insertion maneuver at encounter to transfer from a flyby to an orbiter trajectory about the planet Mars, and subsequent trim maneuver.

The propulsion system is an integrated, pressure-fed, multi-start, fixed thrust, storable bipropellant system. The primary subassemblies are a propellant feed system, a 1334-N (300 lb_f) thrust rocket engine assembly, and the propulsion module structure. The subsystem was capable of being fueled, pressurized, and monitored before installation on the spacecraft.

This document describes the design, testing, fabrication, and problems associated with the development of the Mariner 9 propulsion system. Also covered are the design and operation of the associated ground support equipment used to test and service the propulsion system.

17. Key Words (Selected by Author(s))	18. Distribution Statement
Mariner Mars 1971 Project Propulsion, Liquid	Unclassified -- Unlimited

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price
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